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PRODUCTION PROJECTS DIVISION

INVESTIGATION AND DEVELOPMENT OF
PROPELLANT FEED SYSTEMS FOR MANNED
SPACE VEHICLES (U)

Final Report *T61*

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D. C.

(NASA Contract NAS 7-169)

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AEROJET-GENERAL CORPORATION



INVESTIGATION AND DEVELOPMENT OF
PROPELLANT FEED SYSTEMS FOR MANNED
SPACE VEHICLES (U)

Final Report To
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D. C.

Contract NAS 7-169

Report No. 2735

17 February 1964

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FOREWORD

This report was prepared by the Von Karman Center of the Aerojet-General Corporation, Azusa, California in partial fulfillment of Contract NAS 7-169, "Investigation and Development of Propellant Feed Systems for Space Vehicles." The program was initiated by Mr. Henry Burlage, Jr., Chief, Liquid Propulsion Systems, NASA Headquarters, Washington, D.C. The Headquarters Project Manager was Mr. Frank E. Compitello, and the program was under the technical direction of Mr. Richard N. Porter of the Jet Propulsion Laboratory, Pasadena, California.

The Production Projects Division of the Von Karman Center was responsible for the performance of the program and was directly supervised by Mr. F. W. Childs, III, under the overall direction of Mr. W. J. Flaherty, Manager, Pressurization Systems Department. Assistance was also provided by the following Aerojet personnel:

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The remaining effort under this contract consists of revisions and additions to the "Design Guide for Pressurization System Evaluation-Liquid Propellant Rocket Engines," published as Aerojet-General Corporation Report No. 2334, Volumes I, II, and III, 30 September, 1962, Contract NAS 7-169.

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ABSTRACT

A

Sixteen candidate propellant feed systems were analyzed and evaluated by an objective rating technique for application to a vehicle for (1) a Lunar exploration mission using liquid fluorine/liquid hydrogen (LF_2/LH_2) propellants and, (2) a Mars orbital mission using oxygen difluoride/diborane ($\text{OF}_2/\text{B}_2\text{H}_6$) propellants. Each candidate system was rated on the basis of weight, reliability, volume and cost. A main tank injection (MTI) system was selected for both missions. Operation of the system is based on the injection of hypergolic propellant directly into the main propellant tanks. Because of the need to assume data for many of the sixteen candidate pressurization systems, the selections are satisfactory for preliminary design purposes only. Further analyses and investigations were recommended. Gas generator ignition delay tests, covering a range of mixture ratios from 0.2 to 40, were conducted for the $\text{OF}_2/\text{B}_2\text{H}_6$ reaction. Expulsion tests with small-scale hardware were conducted for subsurface MTI- B_2H_6 injected into OF_2 and top surface MTI- B_2H_6 injected above OF_2 . Results indicate a need for a more sophisticated injector design than that used for $\text{N}_2\text{O}_4/\text{Aerzine-50}$. A method for precise temperature control of the equipment and the propellants is also required.

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I. INTRODUCTION AND SUMMARY

Contract NAS 7-169 was directed toward improvement of the state-of-the-art in the design of propellant pressurization systems for liquid rocket engines which could then be applied to manned space vehicles. The technical effort was divided into the following areas:

A. Analysis and comparison of promising pressurization systems to establish relative merits for two given missions: A lunar mission using liquid fluorine/liquid hydrogen (LF_2/LH_2) propellants and a Mars mission using oxygen difluoride/diborane ($\text{OF}_2/\text{B}_2\text{H}_6$) propellants.

B. Selection of the most promising pressurization system(s) (obtaining any basic design data not readily available).

C. Design verification through testing of the selected system, and reduction of test results to a suitable form for application to system design.

D. Presentation of design data, based upon previous research, analysis and testing in a form easily adaptable to aid in the design and comparison of propellant feed systems.

E. Provision of more readily usable methods of analysis, including the use of computer programs wherever applicable and advantageous, plus compilation of physical-property data as required.

F. Publication of design data generated by this program to supplement that previously published for pressurization system design and comparison.

The last three items above consisted of additions and revisions to Aerojet-General Corporation Report No. 2334, "Design Guide for Pressurization System Evaluation-Liquid Propellant Rocket Engines," published 30 September 1962 under

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I Introduction and Summary (cont.)

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Contract NAS 7-169. The Design Guide includes the information developed to enable pressurization system selection that was initiated under Contract NAS 5-1108. This information is being reviewed, modified, and rewritten to produce a format that will be more convenient to use. The Design Guide was clarified in critical areas and supplemented with additional information where required.

Items A and B above consisted of an evaluation of sixteen candidate pressurization systems for application to two hypothetical missions. This analysis illustrates the use of information available in the Design Guide. A numerical system evaluation technique was applied to sixteen Component Combinations (candidate pressurization systems). Each Component Combination (see Figures 1 through 16) was evaluated both for a Lunar exploration mission with LF_2/LH_2 propellants and for a Mars orbital mission with $\text{OF}_2/\text{B}_2\text{H}_6$ propellants. Each system was rated on the basis of weight, reliability, volume, and cost, plus other factors, such as compatibility.

The rating was made by developing curves that related the importance of the design parameters to nondimensional numbers for a range of values assigned to the design parameters. The numbers denoting the importance of the parameters are called Influence Coefficients. The final rating of each pressurization system is obtained from the product of its Influence Coefficients. The highest product determines the most suitable system.

Cutoff points, such as a maximum allowable weight, were made by giving an Influence Coefficient of zero to any weight greater than the maximum. For systems that were too heavy, the product of the Influence Coefficients was then zero, eliminating that system from further consideration. The system of stored helium, heated by a solid grain within the pressure vessel (Component Combination 16), was used as a reference and 120% of that system's weight was taken as the cutoff point for the Influence Coefficient curves for both missions. The Influence Coefficients used for both missions are shown in Figures 17 and 18. Criteria for selecting the curve shapes are discussed in following sections of this report.

For the Lunar mission, Component Combination 6 (Figure 6) received the highest Influence Product of 153 out of a possible 480. The second highest Influence Product was 148 for Component Combination 10 (Figure 10).

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I Introduction and Summary (cont.)

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Component Combination 6 (Main Tank Injection) obtained the highest Influence Product of 100 out of a possible 480 for the Mars mission. It was followed by Component Combination 11 (Figure 11), with an Influence Product of 77. The operation of the selected system is based on the injection of hypergolic propellant directly into the main propellant tanks. The resulting reaction gases are used to pressurize the tanks.

Summaries of the Influence Coefficients and the design parameters used to obtain them are given for both missions in Tables 1 and 2. The results of the detailed analyses are summarized in Tables 3 through 12.

The selection procedure illustrated in this report is adequate only for preliminary design purposes. Further analyses are recommended, supplemented by experimental work, to obtain certain data that had to be assumed in some instances. It is felt that the system selection for the lunar mission was generally more significant than that of the Mars mission. This is because the lunar mission required several restarts while the Mars mission consisted of a long coast period, followed by a single firing. Since the lunar mission is of greater interest to NASA and industry, the choice of this system is considered more pertinent.

The last area of technical effort (Item C) was directed toward the development of Main Tank Injection (MTI) systems design data through testing. A limited test program was conducted to determine basic MTI performance combinations. Gas-generator ignition-delay tests, covering a wide range of mixture ratios, were conducted to establish possible combustion limits for the $\text{OF}_2/\text{B}_2\text{H}_6$ reaction. Data obtained agreed with results of glassware MTI tests conducted later in the program. The glassware tests physically resembled a sub-surface MTI reaction. Both static pressurization and expulsion tests were conducted with small-scale hardware to investigate sub-surface and above-surface injection of B_2H_6 . Difficulties were encountered in the handling and use of these cryogenics in an MTI system. The results of the test program indicate a need for a more sophisticated injector design than was used. The flow-restricting orifice injector used in this test program was basically the same as that successfully used in previous MTI tests with $\text{N}_2\text{O}_4/\text{Aerzine-50}$ propellants. Future MTI tests with $\text{OF}_2/\text{B}_2\text{H}_6$ will require improved methods for precise temperature control of the equipment and the propellants.

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II. RESULTS AND RECOMMENDATIONS

A. PROGRAM PARAMETERS

An objective, numerical rating technique was used to compare typical main propellant tank pressurization systems for two assigned missions: a Lunar exploration mission using liquid fluorine/liquid hydrogen (LF_2/LH_2) and a Mars orbital mission using oxygen difluoride/diborane ($\text{OF}_2/\text{B}_2\text{H}_6$).

The Lunar exploration mission consists of eight to ten restarts over a ten day period while the Mars orbital mission is, essentially, an extended coast period followed by a single firing for injection into a Mars orbit.

For each mission, sixteen combinations of pressurization system components were compared which are typical of those commonly used or considered. The pressurization systems consisted of the following types based on their sources of energy:

- Stored gas
- Saturated propellant vapor
- Super-heated propellant vapor
- Monopropellant gas generators
- Bipropellant gas generators
- Solid propellant gas generators
- Electro-mechanical expulsion

B. PROGRAM RESULTS

1. For the Mars mission, the Main Tank Injection (MTI) pressurization system (Component Combination 6 - Sequential Injection) obtained the highest point rating of 100.

2. For the Lunar mission, the same system also obtained the highest point rating (153).

3. Ignition times for $\text{OF}_2/\text{B}_2\text{H}_6$ are approximately one-tenth that observed with $\text{N}_2\text{O}_4/\text{Aerzine-50}$ under similar ignition conditions; i.e., in laboratory glass apparatus and also in Aerojet conventional impinging-stream ignition delay apparatus.

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II Results and Recommendations, B (cont.)

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4. Satisfactory MTI pressurization tests of small (1/2 gallon and 1-1/2 gallon) OF_2 tanks by sub-surface and above-surface injection were not obtained in the limited test program conducted. Freezing of the B_2H_6 occurred in one of the two above-surface injection tests and the sub-surface injection test, while overheating to vaporization of the B_2H_6 occurred in the second above-surface test. Malfunctions of the equipment and ruptured tanks were caused by explosion of accumulated unreacted propellant. However, data was obtained that can now be used to design satisfactory injection systems.

5. It was found that compatibility of metal parts (18-8 stainless or equivalent) in contact with OF_2 is best achieved by passivation with F_2 gas at successively higher pressures up to 300 psi. Use of OF_2 gas for passivation, although sometimes satisfactory, is not consistently adequate.

C. RECOMMENDATIONS

It is recommended that an injector be fabricated and tested for the MTI pressurization of $\text{OF}_2/\text{B}_2\text{H}_6$ using the following design principles:

1. Use of "close-coupled" injector system with the injection shut-off point as close as possible to the point of injection.
2. Use of a sufficiently high injection Δp to assure stream injection and to keep the reaction away from the metal surfaces.
3. Suitable control of the injector system temperature (e.g. by heating or jacketing the injector) to assure maintenance of the desired physical state.
4. Use of pulsed-flow-injection and/or gaseous injection for systems requiring low injection flow rates.

III. GENERAL DISCUSSION

A. MISSION CHARACTERISTICS

Pressurization systems were considered for the two assigned missions: a lunar exploration mission using liquid fluorine/liquid hydrogen (LF_2/LH_2) and a Mars orbital mission using oxygen difluoride/diborane ($\text{OF}_2/\text{B}_2\text{H}_6$).

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III General Discussion, A (cont.)

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The primary factor affecting the relative pressurization system weights, for a given propellant combination, is the thrust profile as a function of time, including the length of coast periods and the number of restarts.

The lunar exploration mission consists of eight to ten restarts over a three to ten day period, while the Mars orbital mission consists, basically, of an extended coast period followed by a single firing for injection into a Mars orbit. Therefore, the lunar exploration mission will emphasize systems that have good restart capability while the Mars orbital mission will emphasize efficient pressurization systems for rapid use in non-restart missions.

The impulse requirements for the two missions are compared in Table 1. While the total impulse of the unmanned Mars orbital mission used in this analysis is less than that required for a manned mission, the comparative weights will not be affected significantly.

B. PROPELLANT PERFORMANCE

A cursory survey and comparison of available propellants suitable for advanced space mission was made, including the following:

<u>Oxidizers</u>	<u>Fuels</u>
Oxygen difluoride, OF_2	Diborane, B_2H_6
Liquid fluorine, LF_2	Liquid hydrogen, LH_2
Liquid oxygen, LO_2	
Tetrafluorohydrazine, N_2F_4	

In a discussion with the NASA Program Managers, liquid fluorine/liquid hydrogen was selected for the lunar mission, and oxygen difluoride/diborane was selected for the Mars mission.

Theoretical mission performance values, given in Table 2, were used for the two mission analyses.

C. SYSTEM EVALUATION TECHNIQUE

The system evaluation conducted was based upon the technique developed in Reference 1. This evaluation is an objective, numerical-rating technique that establishes Influence Coefficients independently of system performance evaluation. The Influence Coefficient curves are prepared to reflect the desired propellant pressurization system design; the resulting numerical rating serves as a measure of how closely each candidate system approaches this design. Methods of analysis and significant design values used in analysis are presented for the Lunar mission in Section IV and for the Mars mission in Section V.

1. Selection of Rating Factors

Considering the Mars and Lunar missions, the factors used in rating the systems are weight, volume, cost, and reliability.

The Mars mission rating factor ranges were as follows:

Weight - 1000 lb, or less

Volume - 180 ft³, or less

Hardware cost - \$32,000, or less

Reliability - No range is established because the component combination configurations were established without regard to redundancy, failure mode analysis, or individual component substitutive comparison.

The Lunar mission rating factor ranges were as follows:

Weight - 16,500 lb, or less

Volume - 3175 ft³, or less

Hardware Cost - \$62,000, or less

Reliability - No range is established for the reasons stated above.

The significance of comparing the different component combinations without redundancy is to allow for a comparison of inherent reliability as an independent rating factor. For other missions that do have established reliability requirements, the candidate component combinations should be designed with redundancy

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III General Discussion, C (cont.)

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to raise each system's reliability to the specified value. The redundancy must eliminate reliability as an independent rating factor, and the added components in each component combination must be reflected in the Total Influence ratings. The added hardware will increase the cost, weight and volume and be so indicated in the Influence Coefficient curves of the corresponding rating factors.

An example of designing a system with redundancy to increase system reliability follows:

Table 10 notes that the reliabilities range from 0.1739 to 0.4331 without redundancy. When considering Component Combination 6 with a relief valve and pressure regulator of low reliability, adding redundant components will increase the relief and pressure regulation function reliabilities. Redundancy is added until the component combination reliability reaches a desired level of, say 90%. In like manner, the other component combinations can have their reliabilities increased to a similar level.

The reliability values that have been used throughout this program have been obtained from various governmental and aerospace industry sources such as the IDEP, AGC files and from contractors associated with Aerojet. These reliability data were assumed to be equally applicable to the four propellants (fluorine, hydrogen, oxygen difluoride and diborane) designated for consideration in the pressurization systems evaluation. Insufficient missile and space flight operational data is available to confirm this assumption for fluorine, oxygen difluoride and diborane and is based on experience and observation of comparable reliability requirements achieved by the chemical process and aerospace industries. Other judgment factors also can be included in the system evaluation technique. For example, any anticipated development, feasibility and reliability demonstration costs can also be expressed as Influence Coefficients with appropriate numerical rating values. This step is best taken as an integral part of selecting propellants and precisely establishing the propulsion system characteristics. Because of time and funding limitations for this program, however, peripheral considerations, such as estimating the cost to upgrade an existing pressure regulator to be operationally compatible with oxygen difluoride were not undertaken. It is believed that

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the treatment shown in evaluating the 16 types of pressurization systems and selecting one of these as optimum is adequate from a preliminary design viewpoint. It is essential that confirming and detailed analyses be undertaken beyond the preliminary design and selection shown in the following section. The final design of the pressurization system would then be the result of further analyses of the type illustrated here.

2. Selection of Influence Coefficient Ranges

Of the four rating factors, reliability is the most important for both missions. The Influence Coefficient ranges for both missions are selected as follows:

<u>Rating Factor</u>	<u>Mars Mission</u>	<u>Lunar Mission</u>
Reliability	0.0 - 12.0	0.0 - 12.0
Weight	0.0 - 10.0	0.0 - 10.0
Volume	0.0 - 2.0	0.0 - 2.0
Cost	0.0 - 2.0	0.0 - 2.0

3. Determination of Influence Curves

With the coordinates determined, the shape of the Influence Coefficient curves becomes a function of desired performance; these curves are shown in Figures 17 and 18.

For the Mars mission, the slope of each curve indicates the desirability of a change in the rating factor. A quantitative cutoff is made at the intersection of the slope of the curve and the Influence Coefficient of 1.0. The quantitative cutoff indicates that the rating factor has an influence coefficient of zero beyond that point. The shapes chosen for these curves were, of necessity, somewhat arbitrary. This was because their choice was not preceded by a preliminary vehicle design that would define the desirability of a parameter change in more detail. Therefore, the Influence Coefficient curves reflect only the fact that reliability and weight were the main considerations in this example.

4. Final Evaluation and System Selection

With the rating curves prepared, 16 Component Combinations (candidate pressurization systems) were first compared on the basis of weight; then on

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the basis of reliability, volume, and cost. After calculation of pressurant flow rate and total weight, the weight, reliability, volume and cost of the system components were obtained from the data in Volume III of Reference 2.

The 16 component combinations represent a variety of pressurant and energy sources. These sources include stored gas, saturated propellant vapor, super-heated propellant vapor, liquid monopropellant and bipropellant gas generators, solid propellant gas generators and an electro-mechanical system. Schematics of the 16 component combinations are shown in Figures 1 through 16.

The Mars mission weight comparison is shown in Table 3, and the Lunar mission weight comparisons in Table 4. A preliminary comparison of the weight tables with the maximum allowed weights eliminated Combinations 1, 2, 3, 4, 5, 9, 11, and 12 from the Lunar mission and Combinations 1, 2, 3, 4, 5, 9, 14, and 15 from the Mars mission.

Component Combination 7, the common-ullage Main Tank Injection system, was eliminated for both missions due to the incompatibility of the fuel-rich pressurization gases with the oxidizer.

Liquid hydrogen was not suitable for regenerative jet pump operation. With LH_2 , a jet pump operating at an expected 50% efficiency could only produce a discharge pressure of 35 psia even with the main propellant tank conditioned at its freezing point (25.2°R). Subcooling of the pumped propellant is required to condense the vaporized propellant used for the drive fluid. Without complete condensation before expansion through the jet pump discharge nozzle, the discharge pressure gain cannot be developed. Even at its freezing point, regenerative operation of the LH_2 jet pump could not be attained due to an excessive drive pressure of 140 psia (Reference 3).

Theoretical discharge pressures of 200 psia appear possible with LF_2 . However, its application for a regenerative jet pump was severely limited by the requirement that the propellant be conditioned to within 5°F of its freezing point (Reference 3).

The 300 psia discharge pressure for the Mars mission is obtainable with OF_2 ; however, its application was also limited to subcooled propellant. By

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conditioning OF_2 to within 6°F of its freezing point, theoretical discharge pressures as high as 500 psia may be obtained by a regenerative jet pump operating at 50% efficiency (Reference 3).

The fuel to be used for the Mars mission was B_2H_6 , conditioned to within 6° of its freezing temperature. A regenerative jet pump discharge pressure of 300 psi cannot be reached with the expected efficiency based on calculations made using the computer program developed in Reference 3.

After the elimination of ten component combinations from each mission due to poor performance, incompatibility and high weight, the following six combinations remained: For the Mars mission, Combinations 6, 8, 10, 14, 15, and 16. For the Lunar mission, Combinations 6, 8, 10, 12, and 16. Mission volume comparisons are presented in Tables 5 and 6. Mission cost comparisons are presented in Tables 7 and 8. Mission reliability comparisons are presented in Tables 9 and 10.

Table 11 is a summary of the weight, volume, reliability, and costs of those component combinations being considered for both missions. These values were applied to their respective Influence Coefficient charts. Resulting coefficients are tabulated in Table 12. The numerical rating of each Candidate Combination was determined by the product of the Influence Coefficients. Final Influence Rating also appears in Table 12.

For the Mars Mission, Component Combination 6, the Main Tank Injection system, with a point rating of 100, was selected. Combination 6 has the highest rating for reliability and volume, with cost and weight also having a high rating.

Component Combination 6, the Main Tank Injection system, with a point rating of 153, would be the best system to accomplish the lunar mission. This combination has the highest rating in cost and reliability, while having a high rating in volume and weight. With similar Influence Coefficient values, Component Combination 10, the liquid bipropellant-gas-generator system, produced a rating of 148.

IV. LUNAR MISSION, ANALYTICAL DATA

The methods of analysis and some of the significant design values used for the lunar exploration mission, employing liquid fluorine/liquid hydrogen for propellants, are discussed below for the 16 sets of pressurization systems considered in this analysis.

A. COMPONENT COMBINATION 1 - STORED GAS, REGULATED

1. System Operation

Figure 1 is a schematic of this system. Stored helium is admitted into the main propellant tanks after passing through a pressure regulator. The pressurizing gas is maintained at a constant pressure in the main propellant tanks during thrust chamber operation.

2. Helium Gas

The weight of usable helium gas necessary to expel the propellant was found by applying the perfect gas law to that volume occupied by the propellant; LH_2 volume is taken at $T = 35^\circ\text{R}$ and $P = 200$ psia; LF_2 volume is taken at $T = 150^\circ\text{R}$ and $P = 200$ psia.

The weight of residual helium is taken as 50.2% of usable mass. This percentage is found by applying the perfect gas laws to the initial conditions in the helium bottle of 4000 psia and 40°R and final conditions of 300 psia and an assumed final temperature of 20°R (80% of an isentropic expansion). Perfect gas law calculations included the compressibility factor.

The weight of total stored helium is the sum of usable helium plus residual helium.

3. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2. A safety factor of 2.0 on ultimate tensile strength was used to meet the manned vehicle safety requirements.

Helium tank

Ti 6 AL/4V at -420°F
4000 psia, 487 ft³

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Main fuel tank

17-7 PH at -420°F
300 psia, 2035 ft³

Main oxidizer tank

17-7 PH at -310°F
300 psia, 1082 ft³

Where tank design temperatures other than those shown in Reference 2 (60°F or -300°F) were used, the weight of tank at the Reference 2 temperature was adjusted by the following calculation:

$$W_{\text{Design Temp}}^{\text{T}} = W_{60^{\circ}\text{F}}^{\text{T}} \left(\frac{\text{Strength}}{\text{Density}} \right)_{60^{\circ}\text{F or } -300^{\circ}\text{F}} \frac{\text{Design Temp}}{\text{Design Temp}} \times \text{Safety Factor}$$

or

$$W_{-300^{\circ}\text{F}}^{\text{T}} \left(\frac{\text{Strength}}{\text{Density}} \right)_{\text{Design Temp}}$$

4. Other Components

The weights of other components were determined from the design curves of Reference 2. These component weights are a small portion of the system weight.

B. COMPONENT COMBINATION 2 - STORED GAS, BLOWDOWN

This component combination is not considered for the lunar mission since it does not have restart capabilities.

C. COMPONENT COMBINATION 3 - STORED GAS, BLOWDOWN

The system operation, helium gas requirements, and tankage weights are the same as described for Component Combination 1 above. The total system weight varies because of the changes made in the other components. Component Combination 3 utilizes a solenoid valve and orifice in place of the pressure regulator and solenoid valve of Combination 1.

D. COMPONENT COMBINATION 4 - SATURATED VAPOR (VAPAK)

This combination is not suitable for the lunar mission. The propellant tank pressure required is above the critical pressure of hydrogen.

E. COMPONENT COMBINATION 5 - SATURATED VAPOR (SECONDARY VĀPAK)

1. System Operation

Figure 5 is a schematic of this secondary Vāpak system. Auxiliary stored propellants are vaporized by electrical heating and the heated gas is used to pressure the main propellants. Auxiliary stored liquid hydrogen is used to pressurize the LH_2 fuel and auxiliary stored liquid fluorine is utilized for pressurizing the LF_2 oxidizer.

2. Pressurizing Gases

The weight of GH_2 and GF_2 necessary to expel the propellants was found by applying the perfect gas law to that volume occupied by the propellants for each phase of the mission. The LH_2 volume is taken at -423°F and 200 psia. The LF_2 volume is taken at -307°F and 200 psia. It was assumed that all of the pressurizing vapor condensed during each coast period.

3. Battery

The battery weight was computed from the battery design data of Reference 2. Total battery weight is based on the use of a nickel-cadmium battery with a weight specific energy of 10.5 watt-hours per pound. Watt-hour requirements were determined from the total Btu requirements of the mission. The Btu requirement was calculated by determining the energy needed to convert the auxiliary LH_2 and LF_2 to pressurants GH_2 and GF_2 for each portion of the mission. Auxiliary fluorine is stored as a saturated liquid at -240°F and 200 psia and is vaporized to a gas at -240°F . The auxiliary hydrogen is stored as a liquid at -400°F and 200 psia. It is heated and vaporized to a gas at -360°F .

4. Tankage

All tank weights were computed from the spherical tank design curves of Reference 2. A safety factor of 2.0 on ultimate tensile strength was used.

Auxiliary LH_2

Ti 6AL/4V at -240°F
200 psia, 145 ft³

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Auxiliary LF_2

Ti 6AL/4V at -400°F
200 psia, 33 ft^3

Main fuel tank

17-7 PH at -423°F
200 psia, 1768 ft^3

Main oxidizer tank

17-7 PH at -307°F
200 psia, 1016 ft^3

Where tank design temperature other than those shown in Reference 2 (60°F or -300°) were used, the weight of tanks at the Reference 2 temperature was adjusted by the following calculations:

$$W_{\text{Design Temp}} = W_{60^\circ\text{F}} \text{ or } W_{-300^\circ\text{F}} \frac{\left(\frac{\text{Strength}}{\text{Density}} \right)_{\text{Ratio } 60^\circ\text{F or } -300^\circ\text{F}}}{\left(\frac{\text{Strength}}{\text{Density}} \right)_{\text{Ratio Design Temp.}}} \times \text{Safety Factor}$$

5. Other Components

The weights of other components were determined from design curves of Reference 2. These component weights are small portions of the system weight.

F. COMPONENT COMBINATION 6 - MAIN TANK INJECTION (SEQUENTIAL)

1. System Operation

Stored helium gas is used to inject LF_2 into the main LH_2 propellant tank. The resulting fuel-rich combustion products pressurize the fuel tank. The LH_2 is expelled to the thrust chamber and to the LF_2 propellant tank. The small amount of LH_2 injected into the main LF_2 tank reacts to pressurize the tank with oxidizer-rich gases. Figure 6 is a schematic of this system.

2. Reaction Gases

Fuel- and oxidizer-rich LF_2/LH_2 theoretical combustion products were HF and H_2 , and HF and F_2 , respectively. In both propellant tanks, the HF

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would condense and freeze during any significant coast period, leaving only gaseous H_2 or gaseous F_2 to accomplish any succeeding pressurization.

To compute the weight of reaction products necessary for main propellant pressurization, an "effective molecular weight" was established and a final gas temperature was estimated for substitution into the perfect gas equation. The purpose of the effective molecular weight was to conserve the mass of the combustion reaction, while not conserving the volume of the condensed portions of the combustion products.

The effective molecular weight was calculated by determining the mole fractions for the gas (MF_G) and for the condensed constituents ($MF_{S/L}$). The condensed constituents were then considered as though they were gaseous, so an equation could be derived for the weight of all combustion products in terms of the actual pressurized volume.

The weight of the gas pressurizing a given volume, V , is

$$W_G = \frac{P MF_G V M_G}{R T}$$

and the weight of the condensed constituents pressurizing volume, V , is (if they were gaseous).

$$W_{S/L} = \frac{P MF_{S/L} V M_{S/L}}{R T}$$

where

P = tank pressure

V = tank volume

R = universal gas constant

T = tank temperature

M_G = average molecular weight of gas

$M_{S/L}$ = average molecular weight of condensate (solid liquid)

MF = mole fraction

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Therefore, total combustion product weight is

$$W_T = W_G + W_{S/L} = \frac{P}{R T} (M_F^G V M_G + M_F^{S/L} V M_{S/L})$$

These equations assume that the fraction of the total volume pressurized by each of these two main constituents is proportional to the mole fraction of the constituent. For the gaseous constituent, for example

$$V_G = M_F^G V \text{ or } V = \frac{V_G}{M_F^G}$$

By substituting the latter form for V in both terms of the total weight equation, it becomes

$$W_T = \frac{P V_G}{R T} \left(M_G + \frac{M_F^{S/L}}{M_F^G} M_{S/L} \right)$$

Thus, the effective molecular weight which is required is found because the effective molecular weight

$$M_e = M_G + \frac{M_F^{S/L}}{M_F^G} M_{S/L}$$

is used in the equation

$$W_T = \frac{P V_G M_e}{R T}$$

where the volume pressurized by the gaseous reaction products is the tank volume. Although the above expression for M_e is correct, it may be simplified by deriving an equivalent expression for M_e as follows:

Consider;

$$M_T = \frac{M_F^G M_G + M_F^{S/L} M_{S/L}}{M_F^G + M_F^{S/L}}$$

but,

$$M_F^G + M_F^{S/L} = 1$$

Then, by substitution

$$\frac{M_T}{M_F^G} = M_G + \frac{M_F^{S/L}}{M_F^G} M_{S/L}$$

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It is seen that the right side of the equation is identical with M_e as derived, and the following simplified expression results.

$$M_e = \frac{MT}{MF_G}$$

The effective molecular weight of the reaction products depends upon what products are present which are, in turn, dependent upon the mixture ratio. If the reaction gases originate in a gas generator, the mixture ratio may be closely controlled. With main tank injection, the mixture ratio cannot be known precisely unless experiments are carried out. By changing the design of the injectors, some latitude is open to the experimenters. Experimental work (References 1 and 9) indicates that mixture ratios similar to those commonly associated with fuel-rich and oxidizer-rich gas generators are attainable in main-tank injection systems. Mixture ratios for gas generator pressurization systems are usually determined by finding the lowest molecular weight products commensurate with temperature. Typical gas generators operate within the temperature range of 1500-2000°R. Accordingly, it was assumed that the mixture ratios were the same for gas generator and main tank injection combustion products. Fuel-rich ratios generally run from about 0.1 to 1.0, depending upon the propellant combination. Oxidizer rich ratios can be as low as 20 and as high as 100 or more.

a. Fuel-Rich Reaction

At a mixture ratio of 1.5, the theoretical chemical reaction was $0.9269 \text{ H}_2 + 0.0731 \text{ F}_2 \longrightarrow 0.1462 \text{ HF} + 0.8538 \text{ H}_2$. The average molecular weight of the total combustion products was 4.63; but after allowance for condensing the HF, an effective molecular weight of 5.43 was obtained.

Since only a small portion of the total propellant was expelled during the final engine firing, a final average gas which was 180° above the propellant's normal boiling point was used, i.e., $180^\circ + 37^\circ = 217^\circ\text{R}$.

b. Oxidizer-Rich Reaction

At a mixture ratio of 120, the theoretical reaction was $0.1195 \text{ H}_2 + 0.7609 \text{ F}_2 \longrightarrow 0.2390 \text{ HF} + 0.6414 \text{ F}_2$. The average molecular weight of the total combustion products was 33.1; but after allowance for condensing the

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HF, an effective molecular weight of 45.5 was obtained. Again the final average gas temperature was assumed to be 180° above the propellant's normal boiling point, i.e., $180^{\circ} + 153^{\circ} = 333^{\circ}\text{R}$.

Because of the number of systems under consideration a complete thermodynamic analysis could not be made. Therefore the final gas temperature is a best estimate. Also, the possible effect of heat transfer from the pressurant to the propellant on propellant stratification and external insulation requirements has not been considered.

Following determination of the total reaction products for each tank, the portions of fuel and oxidizer which went into the reactions were found by

$$W_f = \left(\frac{1}{MR+1} \right) W_T$$
$$W_o = \left(\frac{MR}{MR+1} \right) W_T$$

where

W_T = total reaction product weight

W_o = weight of oxidizer into the reaction

W_f = weight of fuel into the reaction

MR = mixture ratio of the reaction (weight oxidizer)/(weight fuel)

The volume of LH_2 that went into the pressurizing reactions was calculated based on the nominal fuel tank volume and then added to the nominal fuel tank volume for another calculation, based on the increased fuel tank volume. The iteration was repeated until the error was less than 1%. A similar iteration was made for the oxidizer tank.

3. Helium Gas

The weight of usable helium gas necessary to inject the LF_2 into the main fuel propellant tank was found by applying the perfect gas law to that volume occupied by the LF_2 at its normal boiling point of LF_2 , and for a pressure of 300 psia.

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From Reference 1, the weight ratio of total stored He to usable He for an isentropic expansion from 4000 psia at room temperature to 400 psia was found to be 0.73. This figure was divided into weight of usable He required and the total stored He weight was obtained.

4. Tankage

All weights were computed from the spherical tankage design curves of Reference 2. A safety factor of 2.0 on ultimate tensile strength was used.

Helium tank

Ti 6 AL/4V at 60°F
4000 psi, 4.01 ft³

Auxiliary LF₂ tank

17-7 PH at -300°F
300 psi, 7.6 ft³

Main fuel tank

17-7 PH at 540°F
250 psi, (1940 + 115) ft³

Main Oxidizer tank

17-7 PH at 90°F
200 psi (1053 + 29) ft³

Where tank design temperatures other than 60°F or -300°F were used, the weight of a tank at 60°F was adjusted by the ratio of strength-to-weight ratios for 60°F and the desired design temperature.

5. Other Components

The weights of other components were determined from the design curves of Reference 2, or by rule-of-thumb estimates. The weights of these components are a small portion of the system weight.

G. COMPONENT COMBINATION 7 - MAIN TANK INJECTION (ULLAGES)

1. System Operation

Stored helium gas is used to inject a small amount of oxidizer into the main fuel tank. Both main propellant tanks are then pressurized by the fuel-rich reaction gases. A filter and/or demister is placed in the hot-gas line between the propellant tanks. Figure 7 is a schematic of this system.

2. Reaction Gases

As described for the previous component combination, the fuel-rich combustion products ($MR = 1.5$) had a theoretical effective molecular weight of 5.43 after condensation of HF.

The final average gas temperature was assumed to be 180° above the IH_2 normal boiling point ($37^\circ + 180^\circ = 217^\circ\text{R}$), since only a small portion of propellant is expelled during the final engine firing. The final average gas temperature was assumed to be the same (217°R) in the oxidizer tank to account for heat loss between the tanks, despite the higher storage temperature of the oxidizer. Again, the final gas temperature is an engineering estimate rather than the result of a detailed thermodynamic analysis.

These values of molecular weight and gas temperature were substituted into the perfect gas law, and the weights of the reactants were determined. The weight of fuel and oxidizer contributing to the reaction was calculated from

$$W_f = \left(\frac{1}{MR + 1} \right) W_t \quad W_o = \left(\frac{MR}{MR + 1} \right) W_t$$

where

W_t = total weight of reactants

W_f = weight of fuel entering reaction

W_o = weight of oxidizer entering reaction

MR = oxidizer weight/fuel weight, 1.5

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Several iterations were made to account for storage of the reacting fuel in the fuel tank. The liquid volume of the reacting fuel was determined and added to the tank volume for each iteration.

3. Helium Gas

The required weight of helium gas was calculated in the same manner as for the previous component calculation.

4. Tankage

All tankage weights were computed from the spherical tankage design curves of Reference 2. A safety factor of 2.0 on ultimate tensile strength was used.

Helium tank

Ti 6Al/4V at 60°F
400 psi, 4.31 ft³

Auxiliary LF₂ tank

17-7 PH at -300°F
300 psi, 11.2 ft³

Main fuel tank

17-7 PH at 860°F
250 psi (1940 + 161) ft³

Main oxidizer tank

17-7 PH at 550°F
200 psi, 1040 ft³

5. Other Components

The weights of the other components were determined from the design curves of Reference 2 or by rule-of-thumb estimates. The weights of these components are a small portion of the system weight.

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H. COMPONENT COMBINATION 8 - STORED GAS (HEATED)

1. System Operation

Figure 8 is a schematic of this system. Stored helium at $T = -420^{\circ}\text{F}$ (40°R) and $P = 4500$ psia is heated in a thrust chamber heat exchanger, and then used to pressurize both main propellant tanks.

2. Helium Gas

The usable helium necessary to expel the main propellant was calculated by the same method used in Component Combination 1, for the Mars mission.

The residual helium necessary was calculated by the same method used in Component Combination 1, for the lunar mission.

3. Tankage

All tank weights were computed from the spherical tank design curves of Reference 2.

Helium tank

Ti 6Al/4V at -230°F
4500 psia, 21 ft³

Main fuel tank

17-7 PH at -423°F
200 psia, 1768 ft³

Main oxidizer tank

17-7 PH at -307°F
200 psia, 1016 ft³

Tank weight adjustments for design temperature and a safety factor of 2.0 on ultimate tensile strength follow the method used in Component Combination 1, for the lunar mission.

4. Other Components

The weights of other components were determined from design curves of Reference 2. These weights are a small portion of the system weight.

I. COMPONENT COMBINATION 9 - ELECTRO-MECHANICALLY ACTUATED BELLOWS TANKS

1. System Operation

Figure 9 is a schematic of this system. Bellows tankage is collapsed by a d-c motor. The motor output is converted to linear movement by a ball screw.

2. Bellows Tanks

Cylindrical bellows tanks were chosen with a length-to-diameter ratio of 2.0 for the fuel tank, and a length-to-diameter ratio of 1.0 for the oxidizer tank. Tank weights were taken from the design curves of Reference 2.

3. Ball Screw

Ball screw weight was taken from the design curves of Reference 2.

4. Other Components

Weights of remaining components were not computed, as tankage and ball-screw weight were obviously excessive.

J. COMPONENT COMBINATION 10 - DUAL BIPROPELLANT GAS GENERATOR

1. System Operation

Stored helium gas is used to pressurize auxiliary fuel and oxidizer tanks which provide propellants to two liquid bipropellant gas generators. The combustion products from the fuel-rich gas generator are used to pressurize the main fuel tank; the oxidizer tank is pressurized by the products of combustion from the oxidizer-rich gas generator. Figure 10 is a schematic of the system.

2. Reaction Gases

The chemical reactions in the gas generators are the same that occur when injecting a small amount of one propellant into the main body of the other propellant. Thus, the discussion of reaction gases for Component Combination 6 using LF_2/LH_2 for the lunar mission applies directly, with the exception of the final gas temperature in the main propellant tanks.

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Since some heat loss is expected between the gas generators and the propellant tanks and the final firing is a small percentage of the total impulse, the final average gas temperatures were assumed to be 150° above the normal boiling point of the propellant being expelled. The assumed final temperatures were 187°R for the fuel tank and 303°R for the oxidizer tank.

As pointed out above, an engineering estimate was made for the final gas temperature rather than a rigorous thermodynamic analysis. Therefore, the final gas temperature must be considered in this light. The effect of heat transfer and mass transfer to the main propellant on propellant stratification and external insulation was not investigated.

After the weights of fuel and oxidizer pressurants were determined from the perfect gas law, the amounts of fuel and oxidizer for each reaction were found by the equations:

$$W_f = \left(\frac{1}{MR + 1} \right) W_t$$

$$W_o = \left(\frac{MR}{MR + 1} \right) W_t$$

where

W_t = Total reaction product weight

W_f = Weight of fuel into the reaction

W_o = Weight of oxidizer into the reaction

MR = Reaction mixture ratio, oxidizer weight/fuel weight

The sum of the fuel weights for both reactions was used to determine the volume of the auxiliary fuel tank, based on the liquid fuel density. The volume of the auxiliary oxidizer tank was obtained in the same manner.

3. Helium Gas

The weight of helium gas required to pressurize both the auxiliary fuel and auxiliary oxidizer tanks was found by applying the perfect gas law to each tank. The propellant volume, normal propellant boiling temperature, and a pressure of 300 psia were used for each tank.

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The density of He at 400 psia and 40°R was subtracted from the density of He at 4000 psia and 40°R to determine the usable density for an isothermal expansion. The weight-ratio of useful to total He was obtained by dividing the useful density by the density at 4000 psia. This weight-ratio was then divided into the required useful He weight to obtain the total weight.

4. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2. A safety factor of 2.0 on ultimate tensile strength was used.

Helium tank

Ti 6Al/4V at -300°F
4000 psia, 32.6 ft^3

Auxiliary LH_2 tank

17-7 PH at -300°F
300 psi, 102 ft^3

Auxiliary LF_2 tank

17-7 PH at -300°F
300 psi, 37.9 ft^3

Main LH_2 tank

17-7 PH at 540°F
200 psi, 1940 ft^3

Main LF_2 tank

17-7 PH at 80°F
200 psi, 1053 ft^3

When tank design temperatures other than 60°F and 300°F were used, the weight of a tank for 60°F was adjusted by multiplying by the strength-to-weight ratio for 60°F , dividing by the strength-to-weight ratio for the design temperature, then multiplying by the safety factor.

5. Other Components

The weights of other components were determined from the design curves of Reference 2 or by rule-of-thumb estimates. The weights of these components are a small portion of the system weight.

K. COMPONENT COMBINATION 11 - SOLID-PROPELLANT GAS GENERATOR

1. System Operation

A solid-propellant grain is fired to generate the pressurizing gas for both main propellant tanks. Restart is accomplished by the sequential ignition of a separate grain for each main engine firing. Figure 11 is a schematic of this system showing only one solid-propellant grain.

2. Reaction Gases

A sodium-azide-base and an ammonium-nitrate-base solid propellant were considered for this system. During each coast period, 97% by volume of the azide combustion products over the liquid hydrogen would condense. For the ammonium nitrate propellant, about 64% of volume of the combustion products would condense over the LH_2 . Due to the numerous restarts and the contamination of the LH_2 by a large quantity of frozen combustion products, this system was not given further consideration. The weight of this system using an ammonium nitrate propellant would be nearly the same as Component Combination 12.

L. COMPONENT COMBINATION 12 - SOLID-PROPELLANT GAS GENERATOR (BLADDER)

1. System Operation

The operation of this system is identical to Component Combination 11 except that the generated gases collapse a bladder to effect positive expulsion of the main propellants. Figure 12 is a schematic of this system.

2. Reaction Gases

As in the previous component combination, 64% by volume of the ammonium nitrate combustion products will condense in the LH_2 tank during each coast period. To account for condensation on seven firings with all outage used in the last firing, the formula for the total effective volume to be pressurized for the fuel tank is:

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$$\begin{aligned} EV_{ft} = & V_1 + (.64V_1 + V_2) + (.64^2V_1 + .64V_2 + V_3) + \dots \\ & \dots + (.64^6V_1 + .64^5V_2 + .64^4V_3 + .64^3V_4 + .64^2V_5 + .64V_6 + V_7) \end{aligned}$$

where

V_1 = Fuel volume expelled during first engine firing

V_2 = Fuel volume expelled during second engine firing, etc.

This equation reduces to:

$$\begin{aligned} EV_{ft} = & 2.65V_1 + 2.59V_2 + 2.48V_3 + 2.31V_4 \\ & + 2.05V_5 + 1.64V_6 + V_7 \end{aligned}$$

The effective volume of the fuel tank was computed to be 4186 ft³.

Since none of the combustion products condense in the oxidizer tank, the effective volume to be pressurized is the actual volume of the oxidizer tank (1053 ft³).

3. Gas Generator

From Reference 2, the weight of the solid-propellant gas generator per cubic foot of pressurized volume was 0.96 lb/ft³. This number was based on a final gas temperature of 580°R, which was higher than could be expected for this mission and propellant combination. Therefore, it was necessary to adjust to the expected gas temperatures.

It was assumed that the final pressurizing gas temperature would be 150°F above the normal boiling point of the main propellant. This provides gas temperatures of 187°R over the LH₂ and 303°R over the LF₂.

The 0.96 lb/ft³ weight-to-volume factor was adjusted by multiplying by 580°R and dividing by the expected final temperature for the propellant tank considered. The resulting weight-to-volume factor was then multiplied by its respective pressurized volume to obtain the gas generator weight attributed to each tank. These came out to be 12,460 lb for the full tank and 1.838 lb for the oxidizer tank.

4. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2. A safety factor of 2.0 on ultimate tensile strength was used.

5. Other Components

The weights of the other components were not considered as they would have been minor and the solid propellant gas generator weight was excessive by itself.

M. COMPONENT COMBINATION 13 - JET PUMP

1. System Operation

This system makes use of the jet pump principle to pressurize the main propellant tanks. In the jet pump cycle, some of the propellant is diverted through a thrust-chamber heat exchanger. Heat transfer to this propellant raises its internal energy. Passing this fluid through a diverging injection nozzle within the jet pump converts the internal energy into kinetic energy, vaporizing the fluid. The main flow from the propellant tank enters the jet pump downstream of this driving nozzle. The high velocity of the drive fluid reduces the pressure within the pump, drawing through the propellant from the tank. The two propellant streams mix and leave the pump through a single diffuser that converts the kinetic energy of the fluid to the high discharge pressure. Most of the propellant then goes to the thrust chamber injector. A certain amount, however, is again diverted to the thrust-chamber heat exchanger for use as driving fluid in the cycle. To start the cycle, helium pressurized start tanks are used. Figure 13 shows the regenerative jet pump system schematic.

2. Jet Pump

Weight, volume, cost and reliability data for the jet pump were taken from the design curves of Reference 2. Performance calculations were obtained from Reference 3.

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3. Start Tanks

The volume of each start tank was assumed to be the volume of its jet pump multiplied by the total number of restarts.

4. Helium Pressurant

The calculation of the total helium required and its usability was made in the same manner as with Component Combination 10.

5. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2. A safety factor of 2.0 on ultimate tensile strength was used.

Helium tank

Ti 6Al/4V at -423°F
4500 psi, 168 ft³

LF₂ start tank

17-7PH at 60°F
200 psi, 12 ft³

LH₂ start tank

17-7PH at 60°F
200 psi, 14 ft³

Main oxidizer tank

17-7PH at 60°F
50 psi, 1056 ft³

Main fuel tank

17-7PH at 60°F
50 psi, 1939 ft³

6. Other Components

The weights of the other components were taken from the design curves of Reference 2 or rule-of-thumb estimates. They are a small portion of the system weight.

N. COMPONENT COMBINATION 14 - STORED GAS (HEATED) AND FUEL-RICH
BIPROPELLANT GAS GENERATOR

1. System Operation

Combustion products of a fuel-rich bipropellant gas generator pressurize the main fuel tank. Heated helium is used to pressurize the main oxidizer tank. Figure 14 is a schematic of this system.

2. Pressurizing Gases

The weight of auxiliary propellants needed to pressurize the fuel was found by referring to Combination 10 for the lunar mission. The weight of heated helium to pressurize the oxidizer was found by referring to Combination 8 for the lunar mission. The amount of residual helium was calculated by the method used in Component Combination 1 for the lunar mission.

3. Tankage

All tank weights were computed from the spherical tank design curves of Reference 2. A safety factor of 2.0 on ultimate tensile strength was used.

Helium tank

Ti 6Al/4V at -420°F
4500 psia, 115 ft³

Auxiliary fuel tank

17-7 PH at -273°F
300 psia, 100 ft³

Auxiliary oxidizer tank

17-7 PH at -157°F
300 psia, 6 ft³

Main fuel tank

17-7 PH at -273°F
200 psia, 1768 ft³

Main oxidizer tank

17-7 PH at -157°F
200 psia, 1016 ft³

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Tank weight adjustments are similar to the method used in Component Combination 1 for the lunar mission.

4. Other Components

The weights of other components were determined from design curves of Reference 2. These weights were a small portion of the system weight.

0. COMPONENT COMBINATION 15 - STORED GAS (HEATED) AND EVAPORATED FUEL

1. System Operation

Heated helium pressurizes the main oxidizer, and auxiliary vaporized fuel pressurizes the main fuel. Figure 15 is a schematic of the system.

2. Pressurizing Gases

The weight of helium required to pressurize the oxidizer was found by referring to Component Combination 8 for the lunar mission. The weight of auxiliary vaporized fuel was found by referring to Component Combination 5 for the lunar mission. Residual helium calculations follow the method outlined for Component Combination 1 for the lunar mission.

3. Tankage

All tank weights were computed from the spherical tank design curves, Reference 2, modified for design temperature and safety factor as noted under Component Combination 1 for the lunar mission.

Helium tank

Ti 6Al/4V at -260°F
4500 psia, 150 ft³

Auxiliary fuel tank

17-7 PH at -423°F
300 psia, 800 ft³

Main fuel tank

17-7 PH at -273°F
200 psia, 1,768 ft³

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Main oxidizer tank

17-7 PH at -157°F
200 psia, 1,016 ft³

P. COMPONENT COMBINATION 16 - STORED GAS (SOLID-PROPELLANT HEATED)

1. System Operation

The system operation is similar to Component Combination 1, except that additional heat and pressurizing gas is supplied by a solid-propellant grain contained within the helium sphere. Figure 16 illustrates this system.

2. Pressurizing Gases

The pressurant requirements were determined by referring to Component Combination 1. It was assumed that the combustion of the solid-propellant grain maintained the residual helium in the sphere at a temperature equal to the storage temperature. It was further assumed that the helium residual mass occupied 95% of the sphere volume.

3. Tankage

All tankage weights were computed from the spherical tankage design curves at Reference 2, as described in Component Combination 1 for the lunar mission.

Helium tank

Ti 6Al/4V at -423°F
4500 psia, 386 ft³

Main fuel tank

17-7 PH at -423°F
200 psia, 1934 ft³

Main oxidizer tank

17-7 PH at -310°F
200 psia, 1082 ft³

4. Other Components

The weights of other components were determined from design curves of Reference 2. These components comprise a small portion of the system weight.

V. MARS MISSION, ANALYTICAL DATA

The methods of analysis and some of the significant design values used for the Mars mission, employing oxygen difluoride/diborane ($\text{OF}_2/\text{B}_2\text{H}_6$) for propellants, are discussed below for the 16 sets of pressurization systems considered in this analysis.

A. COMPONENT COMBINATION 1 - STORED GAS, REGULATED1. System Operation

Helium gas, stored at ambient temperature and high pressure, is used to pressurize both main propellant tanks. Figure 1 is a schematic of this system.

2. Pressurization

Pressurization weight factors, (i.e., dimensionless factors expressing the weight ratio of propellant expelled to pressurizing gas entering the tank) were derived from experimental results obtained on Project Hydra (Reference 4). These results indicated the following: At 520°R inlet temperature with a tank pressure of 90 psia, 1 lb of helium gas would expel approximately 780 lb of liquid oxygen. At a 200°R inlet temperature and the same tank pressure, each pound of helium would expel approximately 400 lb of the propellant. Other experimental data obtained from that program indicated that the weight factors varied linearly within the range of inlet temperatures mentioned above.

The weight of propellant expelled per pound of helium was assumed to vary inversely with the tank pressure. It was also assumed that other propellants with similar properties would behave in the same manner. Therefore, the weight factors from the test data were adjusted to the tank pressure of 300 psia and for the densities of the expelled propellants. The amount of helium necessary to expel the main propellants was determined with the use of these pressurization weight factors.

3. Residual Helium

The amount of residual helium required was calculated by the same method used in Component Combination 1 for the Lunar Mission.

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4. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2, modified for design temperature and safety factor as in Component Combination 1 for the lunar mission.

Helium tank

Ti 6Al/4V at 60°F
4500 psia, 24.8 ft³

Main fuel tank

17-7 PH at 60°F
300 psia, 87 ft³

Main oxidizer tank

17-PH at 60°F
300 psia, 68 ft³

5. Other Components

The weights of other components were determined from the design curves of Reference 2, or by rule-of-thumb estimates. The weights of these components were a small portion of the system weight.

B. COMPONENT COMBINATION 2 - STORED GAS, BLOWDOWN

The system operation, helium gas requirements, and tankage weights are the same as Component Combination 1 above. Component Combination 2 utilizes a squib actuated valve and orifice in place of the pressure regulator and solenoid valve of Combination 1. Combination 2 is shown schematically in Figure 2.

C. COMPONENT COMBINATION 3 - STORED GAS, BLOWDOWN

The system operation, helium gas requirements, and tankage weights are the same as with Component Combinations 1 and 2 above. The total system weight varies by the changes in the other components. Component Combination 3 utilizes a solenoid valve and orifice in place of the pressure regulator and solenoid valve of Combination 1.

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D. COMPONENT COMBINATION 4 - SATURATED VAPOR (VĀPAK)

1. System Operation

During the long prefiring coast, the propellants in the main tanks were assumed to be passively conditioned to a temperature where the vapor pressures are enough for pressurization. During expulsion, the pressure decays, so the initial tank pressure must be somewhat higher than ordinarily required by the thrust chamber despite the use of a lower Δp across the injector.

2. Pressure Decay

Computer calculations of B_2H_6 and OF_2 vapor pressure decay as a function of propellants expelled were made using the computer program described in Reference 5. The OF_2 decay curves, which appear in Reference 2 have been revised on the basis of more complete thermodynamic and physical property data. The revised curves indicate that OF_2 may be used with a pressure decay from 300 psia to 185 psia and expel 95% of the loaded propellant as liquid. The B_2H_6 pressure decay is greater as it drops from 300 psia to 146 psia with only 88% of the loaded propellant expelled as liquid. B_2H_6 may not be stored at higher vapor pressures due to the risk of decomposition during the 200-day coast period. Therefore the Vāpak system was not considered feasible for this application. It should be noted, however, that the Vapak system is most suitable for low chamber pressure applications (below 100 psia).

E. COMPONENT COMBINATION 5 - SATURATED VAPOR (SECONDARY VĀPAK)

1. System Operation

Figure 5 is a schematic of the secondary Vāpak system, which operates in the same manner as Component Combination 5 for the lunar mission.

F. COMPONENT COMBINATION 6 - MAIN TANK INJECTION (SEQUENTIAL)

1. System Operation

The oxidizer, contained in a small auxiliary tank and pressurized by a helium sphere, is injected into the main fuel tank. The fuel-rich products of reaction are used to pressurize this tank. A bypass in the main fuel line feeds

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a small amount of fuel to the main oxidizer tank. The reaction in this tank yields oxidizer-rich products for pressurization. Suitable pressure drops were maintained between the tanks. Figure 6 is a schematic diagram of this system.

2. Reaction Gases

The analytical method for determining the reaction gas constituents and requirements is the same as for Combination 6 for the lunar mission.

a. Fuel-Rich Reaction

At a mixture ratio of 0.2, the theoretical reaction products were $0.6040 \text{ H}_2 + 0.0001 \text{ HF} + 0.0067 \text{ B}_2\text{O}_3 \text{ (l/s)} + 0.0003 \text{ BF} + 0.0135 \text{ BF}_3 + 0.0004 \text{ BOF} + 0.3750 \text{ B}_{(s)}$.

The effective molecular weight of the combustion products was 10.7. The final gas temperature was assumed to be 1365°R (905°F).

b. Oxidizer-Rich Reaction

At a mixture ratio of 120, the theoretical reaction products were $0.2981 \text{ O}_2 + 0.0582 \text{ HF} + 0.4523 \text{ F}_2 + 0.1720 \text{ F} + 0.0194 \text{ BF}_3$. The molecular weight of the products was 32.5. The final average gas temperature was assumed to be 1330°R (870°F).

3. Reaction Gas and Helium

The reaction gas weights and the helium required were found by using the same methods as for Component Combination 6, on the lunar mission.

4. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2, which were adjusted for design temperature and safety factor as for Component Combination 1, lunar mission.

Helium tank

Ti 6Al/4V at 60°F
4000 psi, 0.001 ft^3

Auxiliary OF_2 tank

17-7PH at -115°F
400 psi, 0.026 ft^3

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Main fuel tank

17-7PH at 1100°F
350 psi, 87 ft³

Main oxidizer tank

17-7PH at 710°F
300 psi, 68 ft³

5. Other Components

The weight of other components were determined from the design curves of Reference 2, or by rule-of-thumb estimates. The weights of these components were a small portion of the system weight.

G. COMPONENT COMBINATION 7 - MAIN TANK INJECTION (CONNECTED ULLAGES)

1. System Operation

Stored helium gas is used to inject a small amount of oxidizer into the main fuel tank. Both main propellant tanks are then pressurized by the fuel-rich reaction gases. A filter and/or demister is placed in the hot gas line between the propellant tanks to condense fuel-rich vapors. Figure 7 is a schematic of this system.

2. Reaction Gases

As described for the previous Component Combination, the fuel-rich combustion products (M.R. = 0.2) had an effective molecular weight of 10.7. The final average gas temperature was assumed to be 1330°R.

These values were substituted into the perfect-gas law and weight of the reactions was determined. The weight of fuel and oxidizer contributing to the reaction was calculated from

$$W_f = \left(\frac{1}{M.R. + 1} \right) W_t \qquad W_o = \left(\frac{M.R.}{M.R. + 1} \right) W_t$$

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where

W_t = total weight of reactants

W_f = weight of fuel entering reaction

W_o = weight of oxidizer entering reaction

M.R. = (oxidizer weight)/(fuel weight)

Iterations were made to account for storage of the reacting fuel in the fuel tank. The liquid volume of the reacting fuel was determined and added to the tank volume for each iteration.

3. Helium Gas

The required weight of helium gas was calculated in the same manner as for the previous component combination.

4. Tankage

All tankage weights were computed from the spherical-tankage design curves of Reference 2, as described for Component Combination 1, R₂ lunar mission.

Helium tank

Ti 6Al/4V at 60°F
4000 psi, 0.001 ft³

Auxiliary OF₂ tank

17-7 PH at -115°F
400 psi, 0.06 ft³

Main fuel tank

17-7 PH at 1150°F
350 psi, 87 ft³

Main oxidizer tank

17-7 PH at 1150°F
300 psi, 68 ft³

5. Other Components

The weights of the other components were determined from the design curves of Reference 2, or by rule-of-thumb estimates. The weights of these components were a small portion of the system weight.

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H. COMPONENT COMBINATION 8 - STORED GAS (HEATED)

1. System Operation

Helium stored at -220°F and 4500 psia, was heated in a thrust-chamber heat exchanger and then used to pressurize both main propellant tanks. Figure 8 is a schematic of this system.

2. Helium Gas

The usable helium necessary to expel the main propellant was calculated by the same method used in Component Combination 1, Mars mission. The residual helium necessary was calculated by the same method used in Component Combination 1, lunar mission.

3. Tankage

All tank weights were computed from the spherical tank design curves of Reference 2.

Helium tank

Ti 6Al/4V at -220°F
4500 psia, 21 ft³

Main fuel tank

17-7PH at -134°F
300 psia, 87 ft³

Main oxidizer tank

17-7PH at -228°F
300 psia, 68 ft³

Tank weight adjustments follow the method used in Component Combination 1, lunar mission.

4. Other Components

The weights of the other components were determined from the design curves of Reference 2 and are a small portion of the system weight.

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I. COMPONENT COMBINATION 9 - ELECTROMECHANICALLY ACTUATED BELLOWS TANKS

1. System Operation

Bellows tankage is collapsed by a d-c motor. The motor output is converted to linear movement by a ball screw. Figure 9 is a schematic of this system.

2. Bellows Tanks

Cylindrical bellows tanks were chosen with a length-to-diameter ratio of 2.0 for the fuel tank and of 1.0 for the oxidizer tank. Tank weights were taken from the design curves of Reference 2.

3. Ball Screw

Ball-screw weight was taken from the design curves of Reference 2.

4. Other Components

Weights of remaining components were not computed, as tankage and ball screw weight was obviously excessive.

J. COMPONENT COMBINATION 10 - DUAL BIPROPELLANT GAS GENERATORS

1. System Operation

Figure 10 is a schematic of the dual bipropellant gas-generator system. The combustion products from the fuel-rich gas generator are used to pressurize the main fuel tank, while the oxidizer tank is pressurized by the products of combustion from the oxidizer-rich gas generator. Stored helium is used to pressurize the auxiliary propellant tanks that provide fuel and oxidizer to the gas generators.

2. Reaction Gases

The discussion of reaction gases for Component Combination 6, Mars mission, applies directly, with the exception of the final gas temperature in the main propellant tanks. The final average gas temperatures were assumed to be 540°R for the oxidizer tank and 1000°R for the fuel tank.

Calculations for reaction-gas weights were made by the same method used in Component Combination 10, lunar mission.

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3. Helium Gas

Calculations were made by the same method used in Component Combination 10, lunar mission.

4. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2.

Helium tank

Ti 6Al/4V at 60°F
4000 psia, 10 ft^3

Auxiliary OF_2 tank

17-7 PH at -300°F
300 psia, 192 ft^3

Auxiliary B_2H_6 tank

17-7 PH at -300°F
300 psia, 0.73 ft^3

Main OF_2 tank

17-7 PH at 1000°R
200 psia, 1940 ft^3

Main B_2H_6 tank

17-7 PH at 540°R
200 psia, 1053 ft^3

Tank weight adjustment follows the method used in Component Combination 1, lunar mission.

5. Other Components

The weights of the other components were calculated from the design curves of Reference 2, or by rule-of-thumb estimates. These weights were a small portion of the total system.

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K. COMPONENT COMBINATION 11 - SOLID PROPELLANT GAS GENERATOR

1. System Operation

Both main propellant tanks are pressurized by a solid-propellant gas generator. The reaction gases are passed through a filter before entering the tanks. This system is shown schematically in Figure 11.

2. Solid Gas Generator

From Reference 2, the weight of the solid propellant gas generator was 1.9 lb/ft^3 of pressurized volume. The 1.9 lb/ft^3 assumes a final gas temperature of 580°R , which is higher than could be expected for this combination of cryogenic propellants and mission.

It was assumed that the final pressurizing gas temperature would be 460°R for this solid gas generator, using AG 2838 MOD 1 solid propellant. The 1.9 lb/ft^3 weight-to-volume factor was adjusted by multiplying by 580°R and dividing by the assumed final gas temperature of 460°R . The adjusted weight-to-volume factor was multiplied by the total pressurized volume to obtain the weight of solid-gas generator required for the mission.

3. Tankage

Tank weight design follows the method used in Component Combination 1, lunar mission.

Main fuel tank

17-7 PH at 0°F
300 psia, 87 ft^3

Main oxidizer tank

17-7 PH at 0°F
300 psia, 68 ft^3

4. Other Components

The weights of these components are a small portion of the total system weight. Their design was taken from Reference 2.

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L. COMPONENT COMBINATION 12 - SOLID PROPELLANT GAS GENERATOR (BLADDER)

1. System Operation

The operation of this system is identical to that of Component Combination 11, except that the generated gases expand a bladder to effect positive expulsion of the main propellants. Figure 12 is a schematic of this system.

2. System Weight

The only addition to the total weight of Component Combination 11 is the 0.060-in. wall Teflon bladder that was obtained from the design curves of Reference 2.

M. COMPONENT COMBINATION 13 - JET PUMP

1. System Operation

The operation of this system is exactly the same as that of Component Combination 13 for the lunar mission.

2. Jet Pump

Information for the jet pump was obtained from the design curves of Reference 2.

3. Start Tanks

The volume of each start tank was assumed to be five times the volume of its respective jet pump.

4. Helium Pressurant

The calculation of the total helium required and usability was made in the same manner as Component Combination 10, lunar mission.

5. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2.

Helium tank

Ti 6Al/4V at -223°F
4500 psi, 5.65 ft³

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OF₂ start tank

17-7PH at 60°F
300 psi, 1.0 ft³

B₂H₆ start tank

17-7PH at 60°F
300 psi, 1.0 ft³

Main oxidizer tank

17-7PH at 60°F
50 psi, 68 ft³

Main fuel tank

17-7PH at 60°F
50 psi, 87 ft³

6. Other Components

The weights of the other components were taken from the design curves of Reference 2 or from rule-of-thumb estimates. They were a small portion of the system weight.

N. COMPONENT COMBINATION 14 - STORED GAS (HEATED) AND FUEL-RICH Bi-PROPELLANT GAS GENERATOR

1. System Operation

A fuel-rich bipropellant gas generator pressurizes the fuel tank. Heated helium pressurizes the oxidizer tank. Figure 14 illustrates this system.

2. Pressurizing Gases

The pressurant requirements were found by referring to previous Component Combinations; Combination 8 for the oxidizer tank and Combination 10 for the fuel tank.

3. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2, as described for Component Combination 1, lunar mission.

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Helium tank

Ti 6Al/4V at 60°F
400 psi, 4.7 ft³

Auxiliary fuel tank

17-7 PH at 60°F
500 psi, 1.34 ft³

Auxiliary oxidizer tank

17-7 PH at 60°F
500 psi, 0.07 ft³

Main fuel tank

17-7 PH at 120°F
300 psi, 87 ft³

Main oxidizer tank

17-7 PH at 60°F
300 psi, 68 ft³

4. Other Components

The weights of other components were determined from the design curves of Reference 2, or by rule-of-thumb estimates. The weights of these components were a small portion of the system weight.

0. COMPONENT COMBINATION 15 - STORED GAS (HEATED) AND EVAPORATED FUEL

1. System Operation

A secondary Vapak arrangement pressurizes the fuel tank and heated helium is used for the oxidizer tank. Figure 15 is a schematic of this system.

2. Pressurizing Gases

The pressurant requirements were found by referring to previous Component Combinations; Combination 8 for the oxidizer tank and Combination 5 for the fuel tank.

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3. Tankage

All tank weights were computed from the spherical tankage design curves of Reference 2, as described for Component Combination 1, lunar mission.

Helium tank

Ti 6Al/4V at -228°F
400 psi, 10.85 ft³

Auxiliary fuel tank

17-7 PH at 60°F
4000 psi, 13.2 ft³

Main fuel tank

17-7 PH at 60°F
300 psi, 87 ft³

Main oxidizer tank

17-7 PH at 60°F
300 psi, 68 ft³

4. Other Components

The weights of other components were determined from the design curves of Reference 2, or by rule-of-thumb estimates. The weights of these components were a small portion of the system weight.

P. COMPONENT COMBINATION 16 - STORED GAS (SOLID PROPELLANT HEATED)

1. System Operation

This system operates similarly to the stored helium Component Combination 1, except that additional heat and pressurizing gas is supplied by a solid propellant grain contained within the helium sphere. Figure 16 illustrates this system.

2. Pressurizing Gases

The pressurant requirements were determined by referring to Component Combination 1. It was assumed that the combustion of the solid propel-

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3. Tankage

All tankage weights were computed from the spherical-tankage design curves of Reference 2, as described in Component Combination 1, lunar mission.

Helium tank

Ti 6Al/4V at 60°F₃
4500 psia, 23.7 ft³

Main fuel tank

17-7 PH at 60°F₃
300 psia, 87 ft³

Main oxidizer tank

17-7 PH at 60°F₃
300 psia, 68 ft³

4. Other Components

The weights of other components were determined from design curves of Reference 2. These components comprised a small portion of the system weight.

VI. TEST PROGRAM

The test program objectives were the collection and analysis of basic information on the operational characteristics of a bipropellant gas generator and a Main Tank Injection pressurization system for "shallow cryogenic" propellants. The most important considerations are the density and the temperature of the pressurizing gases produced as well as the characteristics and the amounts of non-gaseous combustion products. Both the ullage gas temperature and its density are controlled by the method of injection (above surface, below surface, etc.). This determines the combustion mixture ratio and, hence, the composition of the gases formed, the amount of heat absorbed by the propellant and the overall weight of combustion products required for pressurization. In the gas generator system, the mixture ratio is set by orificing the fuel and oxidizer feed lines. In the Main Tank Injection system, only subsurface injection can be guaranteed to yield a consistent mixture ratio. Top surface injection tests were conducted to determine if a significant variation in mixture ratio does occur with that mode of injection. Intermediate conditions such as below-surface injection that becomes exposed during the course of the run were also to be evaluated.

The propellants OF_2 and B_2H_6 were chosen as representative "shallow cryogenic" space storable propellants because of the considerable interest exhibited by industry in oxidizers containing fluorine and fuels containing boron.

Additionally, some of the products of the fuel-rich and oxidizer-rich $\text{OF}_2/\text{B}_2\text{H}_6$ reactions appear in the reaction products of many of the advanced propellant combinations that are of interest for space vehicles. Thus, the study of these reaction products can be partially applicable to many other systems.

Theoretical fuel- and oxidizer-rich $\text{OF}_2/\text{B}_2\text{H}_6$ reactions indicate that all three formation processes for non-gaseous reaction products occur. These are the formation of a solid in the reaction gases, the condensation of a gas to a liquid, and the condensation of a gas to a liquid which then freezes.

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VI Test Program (cont.)

Report No. 2735

A. MTI IGNITION DELAY TESTS

The objective of these tests was to study the effect of injection of OF_2 below the surface of liquid B_2H_6 with both materials conditioned to the same temperature (-200°F). This provided a check of Main Tank Injection (MTI) ignition delay against the gas generator ignition delay conditions, simulated with OF_2 at liquid nitrogen temperature and B_2H_6 at dry ice temperature. The MTI tests were conducted in glassware to permit high-speed motion pictures to be taken of the injection process.

1. Apparatus

A schematic diagram of the apparatus is shown in Figure 19. The cameras and electromagnet shown were connected to a sequencing unit. When the firing switch was closed, the sequencing unit started the cameras, and 2 seconds later it turned off the power to the electromagnet.

A photograph of the glass reaction vessel is shown in Figure 20. The main tube, containing the steel ball (0.5 in. dia), was constructed from 20 mm OD glass tubing approximately 200 mm long. The capillary-tipped side tube was constructed from 9 mm OD glass tubing.

Three high-speed motion picture cameras were used to record the tests. An overall view of the test apparatus was taken by a Photosonic camera with nominal speed of 1,000 frames/second. A Fastax camera with a maximum speed of 7,000 frames/second was used to obtain a close-up of the reaction in the lower end of the main tube. A larger Fastax camera with a maximum speed of 14,000 split frames/second (7,000 frames/second) was focused to follow the start and anticipated horizontal travel of the reaction and glass assembly.

2. Procedure

For comparison with the $\text{OF}_2/\text{B}_2\text{H}_6$ test, and to ensure that the sequencing unit was performing properly, a preliminary test was conducted in which N_2O_4 was injected into Aerozine-50 at room temperature. This preliminary test also afforded a check on camera focusing, field of view, and exposure.

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VI Test Program, A (cont.)

Report No. 2735

a. N_2O_4 /Aerazine-50

With the steel ball resting on the capillary tube, N_2O_4 (0.5 ml) was metered into the side tube from a hypodermic syringe, and the tube was manipulated to fill the capillary with liquid. The N_2O_4 was frozen in a dry ice-cooled bath, and the side tube was sealed off at the constriction (Figure 20). When the reaction vessel had warmed to room temperature, the steel ball was raised to a height of about 6 in., by means of a small permanent magnet. Aerazine-50 (10 ml) was metered into the main tube through the stopcock, and the electromagnet was energized to transfer the steel ball from the permanent magnet. The stopcock was left open, and the ball was allowed to drop onto the capillary.

b. $\text{OF}_2/\text{B}_2\text{H}_6$

With the steel ball resting on the capillary tube, the side tube was connected to a small steel cylinder containing OF_2 . To prevent moisture condensation when cooled with liquid nitrogen, the main tube was evacuated. The side tube was then evacuated and the reaction vessel was immersed in liquid nitrogen. The OF_2 was allowed to distill into the side tube until approximately 0.5 ml was condensed, then the side tube was sealed off. The reaction vessel was removed from the liquid nitrogen, the steel ball raised by a permanent magnet and the reaction vessel immersed as quickly as possible in a Dewar flask filled with liquid Freon-14 (boiling point -200°F). The main tube was connected to the diborane cylinder (Figure 19), the lines and main tube evacuated and flushed with nitrogen. Evacuation and flushing were repeated several times to ensure removal of all traces of air. After final system evacuation, diborane (approx. 10 ml) was allowed to distill into the main tube. The electromagnet was energized to transfer the steel ball from the permanent magnet, which was removed remotely.

3. Experimental Resultsa. N_2O_4 /Aerozine-50

Visual observation of the test showed only that a vigorous boiling action occurred in the reaction vessel following the release of the steel ball by the electromagnet. An inspection of the high speed motion pictures was more informative. In the pictures taken at a nominal speed of 7000 frames/second (6000 frames/second actual, as determined by timing marks on film), the ball was seen to break the capillary. The N_2O_4 was injected initially as a liquid and then as a gas into the Aerozine-50.

A blue-gray mixing haze enlarged around the break and changed to a bluish color, which was identified as the reaction flame. It was difficult to determine when the blue area (reaction flame) was first established. It may have initiated by the 5th frame, (0.83 millise), but did not clearly contrast with its surroundings until the 11th frame (1.83 millise).

The N_2O_4 flow reversed momentarily in the 13th frame (2.17 millise) which showed that the reaction pressure was greater than the N_2O_4 vapor pressure (approx. 20 - 30 psia).

By the 15th frame (2.5 millise), the reaction became vigorous enough to start moving the ball back up the tube. The ball then moved up to the top of the tube on a jet of liquid. When the jet and ball settled, the reaction appeared to subside. It is believed that the reaction vessel would have been ruptured had the stopcock at the top of the tube been closed during the test. A frame-by-frame account of the motion picture record is given in Table 13.

b. $\text{OF}_2/\text{B}_2\text{H}_6$

Visual observation of the test disclosed that the reaction vessel and coolant-containing Dewar flask disintegrated almost immediately after the steel ball was released. The high speed motion pictures were not as distinct as they were for the N_2O_4 /Aerozine-50 test, due to the presence of a small amount of frost on either the reaction vessel or the inner wall of the Dewar flask. However, it was possible to see the ball falling, the capillary breaking, the violent reaction that followed.

Liquid OF_2 did not appear to be in the capillary tube. Thus, the observed reaction probably was between gaseous OF_2 and liquid diborane.

The reaction sequence was recorded by both Fastax cameras. The film speed for the reaction closeup was 5,600 frames/second while the reaction and horizontal travel camera operated at a film speed of 12,000 split-frames/second.

For both cameras, no reaction was visible in one frame; but in the next frame, a definite reaction was seen in the straight section at the end of the capillary tube. The intensity of the glow produced by the reaction resembled a neon sign tube with a whitish, slightly blue color. Based on the split-frame camera speed, the reaction was initiated within 0.083 milliseconds.

In both tests, the capillary end of the side tube broke at the bend below the straight end section of the tube. This is the point where the capillary tube is nearest the wall of the larger tube (Figure 20).

After 0.250 millisecond, the reaction in the detached section of the capillary tube faded and the reaction moved down the tube from the break to the base of the large tube. The reaction enlarged within the main tube and at 1.167 millisecond (split frame 14) the ball began to move upward. The test tube began to break between 0.9 and 1.5 millisecond depending on which camera record is used. The inner wall of the Dewar flask began to break between 1.6 and 2.0 millisecond and the outer wall began to break between 1.8 and 2.3 millisecond. The Dewar was shattered and the entire apparatus was expanding by 2 to 2.5 millisecond. The divergence of these times is due to the continued acceleration of the split frame camera. A frame-by-frame description of both camera records is given in Tables 14 and 15.

The travel of the apparatus fragments was followed on the split frame camera. Frame numbers were noted as the fragments passed background distance marks. These distances and the Dewar dimensions were used to determine several average velocities using the split frame pictures.

The average velocities obtained from the split frame camera were:

Dewar inner wall to outer wall	60 meters/second
Dewar outer wall to outer wall plus 1-1/4 in.	19 meters/second
Dewar outer wall to outer wall plus 10 in.	12.6 meters/second

It is evident that the reaction rate did not approach that of a detonation.

Due to the uncertainty of the distance mark to the right of the Dewar and disagreement with the other calculated velocities that 9 meter/second velocity to 1/2 in. beyond the Dewar outer wall was discounted.

From the inner to the outer wall of the Dewar (0.01 meter), the velocity of 60 meters/second was obtained (split frame 24 to 26). From the Dewar outer wall left to the first tape (.032 meter), the velocity of 19 meters/second was obtained (split frame 26 to 46). From the Dewar outer wall right to the only tape (about 0.11 meters), the velocity of 9 meters/second was obtained (split frame 26 to 42). Across 10 in. (0.254 meter) to the left of the Dewar, the velocity of 12.6 meters/second was obtained (240 split frames).

B. GAS-GENERATOR IGNITION DELAY TESTS

This work was carried out to determine the ignition characteristics of the liquid $\text{OF}_2/\text{B}_2\text{H}_6$ system at high and low mixture ratios - i.e., to determine whether smooth ignition and gradual pressure buildup can be achieved at extreme oxidizer/fuel (O/F) ratios that might be experienced with gas generator or Main Tank Injection pressurization reactions. The work was intended to show only that ignition delay was not excessive and was not intended to arrive at a precise determination of ignition delay if it was small.

Ignition delay tests were conducted with O/F mixture ratios ranging from 0.2 to 40; rapid hypergolic ignition, accompanied by generally smooth initial and sustained chamber pressure, was achieved.

1. Apparatus and Procedure

The ignition delay apparatus has been used to determine ignition delays for many liquid bipropellant combinations. It is constructed

from a 1-1/2-in. elbow. One end is fitted with a blowout disk. The other end contains an 0.21-in.-dia nozzle. An exhaust manifold of 2-in. stainless steel tubing is connected to both the blowout disk and the nozzle outlet; the manifold vents to a water scrubber outside the test bay. Single fuel and oxidizer injector elements cause the propellants to impinge at a 45° angle in the chamber. The chamber pressure is measured by a transducer connected to the side of the chamber. The fuel and oxidizer flows are initiated by an Aerojet bipropellant valve equipped with aluminum seats. An extensometer is connected to the valve yoke to record the valve opening and closing and the chamber-pressure trace on an oscillograph.

Cooling jackets were provided around the oxidizer and fuel tanks to liquefy the cryogenic propellants. Cooling troughs were provided around the lines from the bipropellant valve to the injectors. Liquid nitrogen (-195°C) was employed in the OF_2 tank jacket and delivery line trough. Dry ice (-78°C) was used in the diborane tank jacket and delivery line trough.

The diameters of the injector orifices used in these tests were 0.013 and 0.063 in. These injectors were interchanged to shift from a high to a low O/F ratio. Different propellant tank pressures were used to vary the O/F ratios over still wider ranges.

The injectors were calibrated by flowing with water at various pressures. The data from the resulting calibration curves were then used to calculate the mixture ratio (M.R.) at the instant of ignition (when the chamber pressure is still at 0 psig). The actual propellant flows are obtained through correction of the water-flow rates by multiplying them by the square root of the flowing propellant densities. The following propellant densities were used:

	<u>Temp, $^\circ\text{C}$</u>	<u>Density, g/cc</u>
OF_2	-195	1.77
B_2H_6	-78	0.42

Ignition delay times are normally taken as the difference in time (milliseconds) from the midpoint of the bipropellant valve opening trace to the first rise in chamber pressure, as recorded on the oscillograph. In this system, however, the ignition delay times were so short that negative

values were obtained in some cases; therefore, the start of the bipropellant-valve opening was used for the determination of these ignition delay times. This method introduces an uncertainty of about ± 10 millisecc in the ignition delay time.

2. Experimental Results

a. Ignition Delay Data

A total of five test series were run, four of which produced useful data (see Table 16). The first series was conducted with a low O/F ratio (0.19); three tests were made in this series. In every case, smooth ignition was achieved, with no major pressure spikes. The ignition delay times were very short (8 millisecc). The chamber pressures (at steady-state operation) were somewhat lower than expected, probably because the O/F ratio was so far from optimum.

The second series of runs was made with an O/F ratio of about 17. The injector orifices were switched before these runs to obtain the desired mixture ratios without excessive pressure drops. The ignition times were all short (0 to 14 millisecc), but were scattered. On the first run of this series, some air in the combustion chamber may have caused a very short (0-millisecc) ignition delay and a higher-than-average chamber pressure (150 psig); however, ignition was still relatively smooth.

The next series of runs is not reported in Table 16, because a rough start was experienced and the chamber pressure transducer was damaged. No data was obtained and no other damage occurred. These runs were made with an O/F ratio of 50. The pressure-relief blowout disk (set at approximately 800 psig) was not blown. It is believed that the hard start was caused by oxidizer contamination of the fuel injector that resulted from faulty operation of the bipropellant valve. Some difficulty had been experienced in obtaining proper sealing of the pintles with the aluminum seats, and an adjustment had been made before this run.

In the fourth series of runs (4a and 4b), a high O/F ratio (40) was achieved by increasing the oxidizer-tank pressure to

800 psig and decreasing the fuel-tank pressure to 200 psig. The relatively small fuel flow along with the high O/F ratio explains the low chamber pressure obtained (40 psig).

In the last series of runs (3a, 3b, and 3c), the O/F ratio was 30. The ignition time was found to be about 20 millisec, which is still considered to be short. Ignition was smooth and pressure rise slow. The low chamber pressure (25 psig) may have been caused by partial clogging in the fuel injector; if such was the case, the actual O/F ratio would have been higher than the indicated value of 30.

b. Propellants

The fuel used in these runs was commercial-grade diborane obtained from the Callery Chemical Company; reported purity was 95%. The diborane was transferred from the liquid phase in the storage tank to the propellant tank by immersing the propellant tank in liquid nitrogen. The oxidizer used in these tests was commercial-grade OF_2 obtained from the Allied Chemical Company.

c. Oxygen Difluoride Explosion

Before the ignition runs reported above, a small explosion occurred in the oxidizer lines and tank. This difficulty was traced to a faulty check valve that allowed some of the OF_2 to leak into a purge line that had not been cleaned and passivated. The resulting explosion burned through the 1/4-in. stainless steel line, and part of the explosion force propagated back through the lines and ruptured the OF_2 propellant tank.

d. Passivation for Oxygen Difluoride

Stainless-steel lines, tubes, and valves were used for the OF_2 system. The valves had Teflon stem packings and metal-to-metal seats except for the bipropellant valve, which had soft-aluminum seats. All lines and fittings were carefully degreased, pickled, rinsed with Sumco, and dried with N_2 gas. The system was then evacuated for 1 hour with a laboratory vacuum pump. Finally, the system was passivated by bleeding in OF_2 gas and venting as follows:

<u>Bleeding Duration, min</u>	<u>Pressure psig</u>	<u>Remarks</u>
10	10	Followed by venting
10	50	Followed by venting
10	100	-
10	300	-

No heat buildup was noted in any of the lines, and no operating difficulties were experienced with the ignition delay equipment treated in this way.

3. Conclusions

The ignition delay for this propellant combination is short enough to permit satisfactory operation over an extensive mixture-ratio range.

C. PRESSURIZATION TESTS

Pressurization tests were begun after the ignition delay tests had shown the propellants to be adequately hypergolic at or near mixture ratios theoretically existing with Main Tank Injection.

The first test consisted of injecting B_2H_6 into the ullage of a vessel containing OF_2 . The object was to react enough of the two propellants to raise the OF_2 ullage to 300 psig. The system would be left open long enough for the B_2H_6 flow to decrease to zero, with the final ullage pressure becoming equal to the B_2H_6 supply pressure (400 psig). The test system is shown schematically in Figure 21.

A high-pressure cylinder, with a volume of 97.6 in.^3 (1600 cm^3) was used to contain the OF_2 . An ullage of 90% was used to provide a safety factor if the reaction proved unexpectedly energetic. The B_2H_6 fuel was contained in a small bomb with a volume of about 2.13 in.^3 (35 cm^3). Enough B_2H_6 was enclosed to complete the test. The same bipropellant valve used in the ignition delay test was used to initiate this test; B_2H_6 only was passed through the valve and no leaks were evident. All lines and valves of the test equipment were made of stainless steel. An injection orifice with a diameter of 0.004 in. was fabricated in one end of a 1/8-in. AN male union.

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A similar but larger orifice was used in conjunction with a differential pressure gage to obtain the flow rate of B_2H_6 introduced into the OF_2 cylinder. The cylinder was instrumented with a pressure transducer and three thermocouples, located at three different levels.

The equipment was assembled and OF_2 was introduced to the test cylinder by admitting the gas from its storage tank. All portions of the system exposed to OF_2 were thoroughly passivated, as in the ignition delay tests. All portions of the system exposed to B_2H_6 were thoroughly purged with dry nitrogen. The OF_2 was liquefied by surrounding the test cylinder with liquid nitrogen; the level of the liquid nitrogen determined the level of the liquid OF_2 . The B_2H_6 vessel and all lines containing B_2H_6 down to the bi-propellant valve were insulated from heat leakage by a jacket filled with dry ice. The system was pressure-tested for leaks.

The recording instruments were connected to the system and the pressurized diborane line (400 psig) was opened to the OF_2 cylinder. The line was left open for 48 seconds and the following observations made:

1. The differential pressure across the metering orifice was excessive, indicating a clogged orifice or minimal flow. Subsequent examination showed the orifice to be clear.
2. The temperature readings of the thermocouples in the OF_2 cylinder remained stationary.
3. The pressure in the OF_2 cylinder remained subatmospheric.

Approximately 2 minutes after the B_2H_6 feed line was closed, three explosions occurred in rapid succession and numerous small popping noises followed. The OF_2 cylinder was demolished, and the bottom was blown out of the nitrogen-bath bucket. The portion of the system above the OF_2 vessel remained intact and unharmed; apparently, all of the OF_2 was consumed in the reactions because no toxic odors were present afterward.

It is theorized that the small (0.004-in.) injection orifice became clogged or was fouled, which prevented direct impingement of the B_2H_6 upon the oxidizer surface. If the orifice had become clogged, the residual pressure of the fuel above it could have cleared it, even after the propellant

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valve was closed. If the orifice was fouled and partially clogged, considerable fuel (up to 1 gram) might have built up in the upper tank area instead of impinging against the surface of the oxidizer. The low vapor pressure of the oxidizer, combined with the probable pressure of some inert nitrogen gas from purges make the second situation more likely and within reason. Subsequent examination showed that the orifice was clear, indicating clogging due to freezing of the diborane.

The data below was taken from the recording instruments that were activated during the 49-second duration of the experiment; all traces remained constant during the recording period.

Pressure in OF_2 vessel	2.7 psia
Temperature of OF_2 liquid	-261°F
Temperature of OF_2 vapor at midpoint of vessel	-212°F
Temperature of OF_2 vapor at top of vessel	-70°F
ΔP reading, indicating fuel flow rate	Off scale

D. EXPULSION TESTS

1. Subsurface MTI - B_2H_6 into OF_2

a. General Discussion

The first expulsion test consisted of injecting diborane under the surface of oxygen difluoride. The gases generated by the hypergolic reaction provides the energy for expulsion. A schematic diagram of the system is shown in Figure 22 and a photograph of the test apparatus is included as Figure 23.

The test system was constructed on a tubular framework, shielded by 1/4-in. steel plate. Provisions were made for either remote or reach-rod operation of all system valves. The legend on Figure 22 shows the type of seats used in each valve. Soft copper seats were used wherever the

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system was exposed to liquid oxygen difluoride. In the fuel side, and in areas where gaseous oxygen difluoride would be present, Kel-F or teflon seats were used.

During assembly, system characteristics were determined for the first subsurface injection run. Orifice calibrations were completed for both the fuel restriction (injector) and fuel measuring orifices. The orifice diameters selected were 0.010 and 0.0135 in. respectively. The orifices were chosen so the metering orifice would give a pressure drop of approximately 25 psi and the injection orifice would function (during steady state operation) with a drop of 50 psi. Because of the question of orifice clogging or fouling in the first pressurization test, it was decided to minimize the problem by designing the test program around the use of larger orifices. A minimum diameter of 0.008 in. was set for the flow-limiting orifice for the MTI injected propellant. This decision resulted in several test plan changes. No significant difficulty with plugged orifices was encountered during calibration. The orifice selected for use at the point of injection was electronically drilled and produced better flow characteristics than two other types tried. A 0.182-in. orifice was selected for the expulsion flow regulating orifice to provide an expulsion time for 1.3 gal. of OF_2 of 10 seconds. A total quantity of 8.14 cc of fuel was to be injected, which is relatively large compared to the fuel feed line hold up (free volume) of 2.40 cc, excluding a filter which was later installed. The internal volume of the filter was determined to be 12 cc, thereby requiring a total fuel consumption of 22.54 cc (plus purge).

The B_2H_6 tank was temperature conditioned with dry ice and the OF_2 with liquid nitrogen. No single fluid was found that was compatible with both propellants and had an intermediate boiling point. Liquid nitrogen would have frozen the B_2H_6 and the vapor pressure of OF_2 would have been excessive at dry ice temperature. Therefore, the liquid nitrogen bath was blocked off so it did not come within 1-1/2 in. of the B_2H_6 injector and the injection line was resistance heated by passing a current from a welding machine through it.

The system was designed to begin pressurization with 350 psig in the fuel cylinder and 200 psig in the pressurization tank. The

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initial fuel feed rate was designed to be 108 cc/min, dropping to 64 cc/min as the expulsion tank pressure reached 300 psig. The expulsion was to be controlled by means of a tank pressurization switch that would signal the expulsion line valve to open when expulsion tank pressure reached 300 psig.

Provision for sampling the expelled liquid after the completion of the run was made. As shown in the schematic diagram, a 350 cc sampling vessel (T_4) was submerged in liquid nitrogen. By shutting Valves 16, 17, 18, and 22, the sample bottle might be detached from the system and its contents analyzed at some future time.

The test system was passivated with OF_2 gas for a period of 18 hours at 75 psig in preparation for the first test.

b. Test No. 1

Preparation was made to conduct the first test (B_2H_6 into OF_2 subsurface) on 23 May 1963. The OF_2 expulsion tank, with the exception of a 3 in. dia area around the injector and the catch tank, were pre-chilled with liquid nitrogen. The B_2H_6 tank, pre-chilled with dry ice, was installed in the system. Filling of the system with OF_2 was started with OF_2 tank pressure maintained at 50 psig. After five minutes, the fill pressure was increased to 75 psig. Approximately 30 sec elapsed, after the pressure was increased, when an explosion occurred in the test system. Servicing was stopped immediately.

After a waiting period of about 30 min, the system was inspected for damage. It was observed that a failure had occurred in the vicinity of the Fischer-Porter flow meter and the 1 in. Annin expulsion valve. Burned insulation was found in this immediate area. The system was then purged with helium.

Removal of the Annin expulsion valve revealed that the pintle had been bent off the seat. Apparently, a local reaction had occurred in the threaded area of the union connection at the valve inlet. A groove was burned across the threads and up the face of the hexagon on the union at this point. The force of the reaction had sheared the rotor blades

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off the hub of the flow meter located about 10 in. upstream of the valve. It is estimated that about 1 lb of OF_2 was involved in the reaction.

It was decided that the OF_2 passivation had been incomplete and when liquid OF_2 entered the valve it had probably reacted with an impurity such as thread lubricant or an unpassivated area. It was further decided that future passivation would be accomplished using gaseous fluorine instead of OF_2 to provide certainty of complete passivation. In addition, the system passivation pressure would be increased to 300 psig before venting the passivation gas.

c. Test No. 2

Before Test No. 2 (D270-LQ-1) the system was assembled and passivated with fluorine gas. The system was pressurized to 75 psig and remained in a pressurized condition over the weekend (approximately 64 hours). The system pressure was increased to 300 psig, allowed to stand for ten minutes, purged, and set up for a test. The main tank and expulsion tank jackets were filled with LN_2 and the fuel tank was installed and packed with dry ice. The OF_2 tank was serviced with 16 lb of OF_2 .

On June 4, 1963, an attempt was made to inject B_2H_6 into the OF_2 main tank at a sub-surface level. There was no evidence of B_2H_6 injection. Several attempts were made by opening and closing the start valve. The B_2H_6 bleed valve was actuated and from the ΔP transducer it was apparent that flow was occurring and therefore the system was operating properly upstream of the start valve. The start valve was found to be operating properly by observing the valve trace on the oscillograph. The injector orifice was known to be open immediately before attempting B_2H_6 injection, because the OF_2 tank pressure was increased by helium flow from the purge system through the injection orifice. It was felt that the main injection orifice and/or line was blocked due to frozen B_2H_6 . To determine if the injection orifice or line between the start valve and the OF_2 tank was frozen, the helium purge was actuated. At this time the OF_2 tank was pressurized to approximately 300 psig. The helium pressure was increased to 500 psig and no pressure rise in the tank was noticed, which verified the previous assumption of frozen B_2H_6 in the injection system.

A welding machine was attached to the line containing the injection orifice. It was decided to provide heating pulses of increasing duration to the injector by manually turning the welding machine on and off. During the third attempt to heat the orifice, the line melted and a fire resulted. The water deluge was initiated and remained on overnight. A review of the damage disclosed the following: on the fuel side, the orifice plate was melted, the start valve was damaged and the seats in the Marotta valves were melted. All damaged lines and hardware were removed and repaired. The electrical harness was replaced. The oxidizer side of the system remained undamaged throughout the entire series of events.

2. Above-Surface MTI - B_2H_6 into OF_2

a. Test Setup

The basic system used for the MTI subsurface OF_2 expulsion was modified and assembled for the above-surface injection of B_2H_6 into OF_2 . A schematic of this system (not to scale) is shown in Figure 24. A gas sampling bottle was installed in addition to the expelled-liquid sampling apparatus. A heating jacket was installed to prevent freezing of the B_2H_6 up to or in the injector orifice.

Again, it was planned to use LN_2 to temperature condition the OF_2 and dry ice to temperature condition the B_2H_6 . This was because LN_2 would freeze the B_2H_6 and an OF_2 vapor pressure of 440 psia was considered excessive when conditioned with dry ice. Temperature conditioning the large OF_2 tank with dry ice was also felt to be hazardous since repacking the OF_2 tank in dry ice would have been a manual operation.

The heating jacket for the injector line consisted of a piece of tubing about 1 in. dia by 1 in. long, into which hot nitrogen was introduced. It was placed around the 1/8-in. dia line leading to the injector orifice and served as a baffle to direct the flow of heated nitrogen. Three iron-constantan thermocouples were welded to the recessed face of the tank end-plate, next to the 1/8-in. union containing the injector orifice. The injector and heating jacket installation are shown to scale in Figure 25. During the installation of the 1/8-in. union containing the injector orifice,

the union broke, leaving the threaded portion continuing into the injector installed in the tank. The hexnut face was placed flush on the tank end-plate and was welded to the tank around the hexnut perimeter while in contact with, and center on, the installed portion by a short piece of wire.

The injector orifice diameter for this test was 0.008 in. and B_2H_6 flow was monitored using a 0.010-in. ΔP orifice. Both orifices were electronically drilled through the end of a 1/8 in. AN union that was welded shut. A steady-state B_2H_6 flow rate of 7.3×10^{-4} lb/sec was expected with these orifices. Based on a mixture ratio of 120, a combustion product molecular weight of 39, an average gas temperature of $1300^\circ R$, and a specific gravity of 1.77 at $-320^\circ F$ for the OF_2 , a weight flow rate of 11.6 lb/sec was calculated for the OF_2 at a tank pressure of 300 psi. This would result in complete expulsion of the OF_2 in 2.3 seconds; however, a timer was set to terminate the expulsion at 2.0 seconds so a sample of the pressurizing gas could be taken. Also, after a cool down period, a sample of the unexpelled OF_2 could be taken that would provide an indication of possible contamination of the liquid.

When the system was first passivated, the OF_2 and the B_2H_6 vent valves showed leakage past the valve stem packings. These valves, 14 and 6 respectively of Figure 24 were removed and repaired. The system was reassembled, leak-tested and passivated with fluorine gas.

The system was pressurized to 75 psig overnight. At 8:00 AM the following day, there was still 60 psig present, indicating a tight system. The system was vented and purged with helium gas. The tank jackets were cooled down. At this point, it was felt that before testing, a complete passivation could be proven by servicing 2 lb of OF_2 into the main tank and subjecting the system to operating pressures to determine if any reaction would result. The OF_2 was loaded and the system was pressurized to 300 psig; no reactions were experienced. The system was vented and the OF_2 was expelled into the charcoal scrubber. The B_2H_6 tank was installed, packed with dry ice and pressurized with helium to 361 psig. The lines up to Valve 6 were filled with B_2H_6 by opening Valve 13 (Figure 24) for a few seconds. The main and

expulsion tank jackets were filled with LN_2 and 265 lb of OF_2 were loaded into the main tank. The injector orifice purge (helium) was increased from 150 psig to 300 psig and the tank ullage was pre-pressurized to about 300 psig. The injector purge was started before loading OF_2 and continued until the test was completed to prevent OF_2 from entering the injector before the test. The nitrogen temperature was set for 300°F and was estimated to be 200°F at the heating jacket.

b. Test Results

On June 21, 1963, the first above-surface injection of B_2H_6 into OF_2 , was conducted (D-270-LQ-2). Upon actuation of the fire switch (FS-1), the valve that starts B_2H_6 flow opened (Valve 6, Figure 24). The pressure remained steady at 311 psig in the OF_2 tank. After 11 seconds the fire switch was deactivated (FS-2) and the valve closed. Following a lapse of 21 seconds, the second attempt was made to inject B_2H_6 ; this time the fire switch was activated for 25 seconds. The OF_2 tank pressure remained constant throughout the second injection attempt, but 40 millisecc after deactivation of the fire switch (FS-2) a reaction occurred in the OF_2 tank. The bolts holding the flat end flange to the top of the OF_2 tank yielded, allowing OF_2 to escape across the flange interface. A fire resulted, damaging the system. The system vented, but could not be inspected until the following day.

Disassembly showed that the circumferential gasket seating grooves on the flange were eroded by the radial flow of the OF_2 and none of the copper gasket remained. A comparator check of the 1-in. dia Type 347 stainless steel tank bolts revealed that they had been stressed beyond the yield point of the material. A permanent set of 0.100 in. was found in a length of approximately 2 in.

Inspection of the injector orifice showed that it had eroded from 0.008 in. dia to approximately 0.020 in. dia. The face of the flat end-flange appeared clean with the exception of the conical recess expanding from the injector fitting to the interior face of the end-flange. This surface and the end of the injector were definitely blackened by the reaction (see Figure 26).

A residue, with the appearance of a granular metallic substance, was found in the bottom of the OF_2 tank and in the OF_2 exit tube below the tank. Samples of the material were obtained and analyzed on a mass spectrometer. A copy of the analysis is presented in Table 17, and indicated that the major constituents were stainless steel.

The oscillograph trace showed that the B_2H_6 flow rate jumped to a peak of 16×10^{-4} lb/second but then fell to zero or a very low value within 0.64 second of FS-1 closure. At the second FS-1 activation, the B_2H_6 flow rate jumped to a peak of 16×10^{-4} lb/second. As before, the flow rate dropped to a zero value within 0.59 second. The B_2H_6 flow rate is plotted as a function of time in Figure 27. The pressures and temperatures recorded during the test are tabulated in Table 18. The locations of these pressures and temperatures are indicated in Figures 24 and 25.

c. Test Analysis

To understand what did occur during this test, the test system and data should be examined in more detail. There were three possible conditions at the 0.008 in.-dia injector orifice: completely plugged, partially clogged or not obstructed at all. It is believed that the last case existed.

If the injector were completely plugged, the overall compression of the helium in the 3.75 cc volume of injector lines (bottom of Table 18) from 311 psig to 361 psig ($\Delta P = 50$ psi) would have required flow of about 0.5 cc of liquid B_2H_6 . From Table 19, this corresponds with the recorded flow volumes of 0.544 cc and 0.428 cc for a B_2H_6 specific gravity of 0.42. There are two factors which this hypothesis does not explain. If the injector orifice were completely plugged, there would not have been any flow of the liquid B_2H_6 upon opening of the valve for the second time. Also, after compression of the helium and a moment to damp out the B_2H_6 flow, the flow rate would be expected to fall off at nearly the same rate it increased when the valve was opened. This would be the case unless some flow was occurring at the injector.

In the cases of the partially clogged and unobstructed orifices, the helium would be compressed from 311 psi, but initially to only 331 psi ($\Delta P = 20$ psi), as the pressure drop across the flow measuring orifice

is 30 psi for a B_2H_6 flow rate of 16×10^{-4} lb/second. This is because the sum of the pressure drops for the flow measuring orifice and the injector orifice must stay a constant 50 psi (Table 15). During the second phase of helium compression, the helium pressure increased from 331 psi to 361 psi ($\Delta P = 30$ psi) as the B_2H_6 flow rate (flow orifice pressure drop) fell to an indicated value of zero. The end of these two phases of helium compression correspond to times of 0.04 and 0.64 second for the initial flow and 31.94 and 32.45 second for the final flow, Figure 27. The total B_2H_6 flow (area under curve, Figure 27) in the second phase is considerably more than would be expected by comparing the change in helium pressure of 30 psi for the second phase to that of 20 psi for the first phase. The B_2H_6 flow in excess of this 1.5:1 ratio must be attributed to flow through the injector orifice and into the OF_2 tank. This shows that flow through the injector did exist but does not indicate whether there was a partial clogging of the injector orifice.

At a B_2H_6 flow rate of 16×10^{-4} lb/second the pressure drop across the injector orifice was 20 psi. This pressure drop results in calculated injector flow rate of 4.16×10^{-5} lb/second for helium, assuming a maximum temperature of $200^\circ F$. The flow rate of helium equivalent to the volumetric flow rate of B_2H_6 at 16×10^{-4} lb/second is only 1.18×10^{-5} lb/second. Assuming that this difference is due to only a partial clogging of the injector, a comparison of these values shows that at least 28% of the injector area was open. However, there are other factors that are more likely to have caused the flow rate of B_2H_6 to drop.

Calculations were made to determine the B_2H_6 flow rate based on three different conditions of injecting gases into the OF_2 tank at temperatures from $30^\circ F$ to $200^\circ F$. This temperature range was chosen because a temperature of $+30^\circ F$ was required to cause the vaporization of the B_2H_6 and it seemed unlikely that a temperature greater than $200^\circ F$ could be reached with the heating method used. The three conditions considered were: injection of helium gas without any vaporization of B_2H_6 , injection of helium gas with a stationary B_2H_6 vaporization interface, and the injection of B_2H_6 vapor with a stationary B_2H_6 vaporization interface.

The injection of helium without any B_2H_6 vaporization would result in a recorded B_2H_6 flow rate between 19.0 and 20.5×10^{-4} lb/second for injection gas temperatures within the 30 to $200^\circ F$ range. The introduction of liquid B_2H_6 into an injector tube pressurized with helium gas will result in the vaporization of B_2H_6 as it tries to reach vapor pressure equilibrium. This vaporization could easily begin within 0.020 second after introduction of the B_2H_6 . The vaporization (without any temperature increase) of 38% of the B_2H_6 would result in the recorded B_2H_6 flow rate peak of 16×10^{-4} lb/second. Any heating of the B_2H_6 liquid or vapor would reduce this percentage that would have to be vaporized.

The recorded B_2H_6 flow rate would drop to 4.5×10^{-4} lb/second for the injection of helium in the 30 - $200^\circ F$ range when 100% of the B_2H_6 is vaporized at a stationary interface. Another drop in recorded B_2H_6 flow rate would occur when the last of the helium is injected and injection of B_2H_6 vapor begins. Assuming a stationary vaporization interface, the liquid B_2H_6 flow rate should vary from 0.73 to 0.84×10^{-4} lb/second for injection vapor temperatures from $200^\circ F$ down to $30^\circ F$, respectively. A minimum temperature of $30^\circ F$ is required because the vapor pressure of the B_2H_6 must be above the total pressure in the tubing for this condition to occur.

From the flow data recorded in Figure 27, the case of injecting B_2H_6 vapor does not appear feasible, because a zero value for liquid B_2H_6 flow was recorded after each short initial flow. The conflicting factor is that the minimum usable flow orifice size of 0.010 in. dia. does not provide enough pressure drop to detect a flow rate of less than 1.3×10^{-4} lb/second. This rate of flow corresponds to 0.020 in. on the oscillograph record and is indistinguishable from instrumentation zero shift. Therefore, an unrecorded liquid B_2H_6 flow rate of 0.84×10^{-4} lb/second could have existed for 36.01 seconds during the two "zero flow" portions of the test.

In summary, it is believed that the following sequence of events were initiated with each opening of the B_2H_6 flow valve. The helium pressure in the injector tubing began to rise and the entering B_2H_6 began to vaporize immediately after opening of the B_2H_6 flow valve (FS-1). A peak

B_2H_6 liquid flow was reached at 16×10^{-14} lb/second due to slightly less than 38% of the B_2H_6 vaporizing at that time. Meanwhile, helium was flowing through the injector orifice. Helium compression and flow continued as the rate of B_2H_6 vaporization continued increasing to 100% of the B_2H_6 flow (0.040-0.35 second). During the same period, B_2H_6 vapor began to reach the injector and the injected gas composition gradually changed from helium to B_2H_6 by 0.63 second. At this time, the liquid B_2H_6 flow rate became undetectable because the pressure drop through the flow measuring orifice had dropped to less than 0.2 psi. This condition continued until the B_2H_6 flow valve was closed (FS-2). Through diffusion and/or the injector helium purge, the B_2H_6 concentration was reduced. This left the system ready to repeat the same sequence upon second actuation of FS-1.

The preceding explanation holds true, even if only 28% of the injector orifice was clear during this entire process; however, some numbers would be changed. The possibility of maintaining an obstruction in 72% of the injector orifice area is discounted for the following reasons:

It is unlikely that the obstruction would remain, blocking a constant percentage of the injector orifice during two injection attempts. Similar measurements were made while injecting N_2O_4 on another program of 12 tests. On every occasion that clogging of the 0.006-in. dia injector orifice occurred, it was very erratic and showed up quite distinctly on the oscillograph record of the flow metering orifice. Since the flow metering orifice was more than twice the diameter of the injector orifice, it was concluded that the clogging occurred at the injector orifice.

The injection of B_2H_6 vapor probably resulted in an explosive mixture that burned to detonation. A small stream of B_2H_6 vapor above $30^\circ F$ (approx. 0.8×10^{-4} lb/second) was injected into the 4.5 in. deep ullage of OF_2 vapor at 0.1 - 1.0 psia, diluted by helium at 326 psia. As the B_2H_6 vapor diffused, it may have condensed and frozen. After some accumulation of B_2H_6 , a reaction occurred and propagated up through the explosive mixture of OF_2 and B_2H_6 diluted by helium. The velocity of the flame front was either high enough to create a shock wave, or the reaction occurred at a very high

mixture ratio (about 120:1 or more). The following data and calculations were used to reach this conclusion:

It is estimated that the temperature near the top of the ullage was about -150 to -200°F . TOT-1 indicated -276°F , 2 in. from the top of the ullage thermocouple (Figure 25), and the OF_2 was at or slightly above -310°F according to TOT-2. Thus, the temperature was below the normal freezing point of B_2H_6 (-264.8°F) 2.5 in. above the OF_2 surface.

To obtain a permanent strain of 0.050 in./in. in the Type 347 stainless steel bolts securing the top of the OF_2 tank, a maximum stress of 61,500 psi is required. This assumes that the normal stress-strain curve for Type 347 may be used for impact loading situations. An impact load that jumped from zero to some constant value was assumed. This value was shifted while comparing its difference between the stress of the stress-strain curve until the gains and losses in kinetic energy were balanced, obtaining an average impact stress. This method is described in Reference 6. From the pressure area (6 in. dia) for the tank flange (held on by eight 1 in. dia coarse thread bolts), this average impact stress of 46,000 psi converts to an average impact pressure of 2,900 psia in the OF_2 tank.

Assuming the ullage was at a uniform temperature of -276°F (TOT-1), the pressures generated by reacting all of the B_2H_6 from the recorded flow and the maximum unrecorded flow ($+30^{\circ}\text{F}$) were calculated. Four cases were considered; zero and complete heat transfer mixing of the reaction gases with the helium were made for reaction mixture ratios of 3 and 120. It was assumed that there were no heat losses to the tank walls or the liquid OF_2 surface. The tank pressures achieved for each case are given in Table 20.

In none of the cases considered did the tank pressure come within 2600 psi of the stress required to cause the permanent strain of 0.050 in./in. in the tank flange bolts. However, if the stress-strain curve for Type 347 stainless steel (with impact loading) is less than 65% of normal loading rates, the permanent strain on the bolts could have been generated without a reaction shockwave.

The following alternate explanation is also presented:

The flame started from the ignition point and moved away from it with increasing speed. The unburned mixture was forced away by the expansion of the burning gas causing a rise in pressure and temperature of the unburned gases. This increased the speed of the flame and thus increased the strength of the compression wave propagated. The temperature of the compressed mixture, seen by the reaction front, increased as the shock wave outdistanced the reaction front. This is due to the increased contact time from frictional heating of the compressed mixture against the tank walls, after passage of the shock (but before reaction). The increased temperature of the compressed mixture caused a corresponding increase in the velocity of the reaction front. When the velocity of the reaction surpassed that of the shock, it entered the region where no steady condition was possible until a steady detonation wave was established. This phenomena is described as "burning to detonation" (Reference 7). The change in flame speed, caused by crossing this unstable region, to a detonation wave is usually quite significant. For example, the flame speed for an acetylene and oxygen mixture would increase from Mach 1.5 to Mach 8.7 (Reference 7).

E. COMPATIBILITY EXPERIENCE

Experience gained during the performance of $\text{OF}_2/\text{B}_2\text{H}_6$ pressurization and expulsion tests, as well as direct compatibility tests, agrees substantially with previously published works. Principal observations were as follows:

1. The fuel (diborane) posed no compatibility problems
2. There was a very great difference in the reactivity of the oxygen difluoride liquid and gas. Compatibility tests were carried out using the gas. As shown in Table 21, no reactions were noted. Similar experiments performed with fluorine gas show that it is much more reactive than oxygen difluoride gas (Reference 8). Before the first expulsion test, there was a small explosion in the liquid expulsion line, possibly caused by the reaction of liquid oxygen difluoride with thread lubricating compound or some other impurity in the area of the expulsion valve. Care had been taken to avoid this

type of problem, because the system had been thoroughly passivated with oxygen difluoride gas. Since the probable reason for this mishap was the lower passivation qualities of the oxygen difluoride gas, the more reactive (as verified by Reference B) fluorine gas was substituted as the passivating agent. No further problems of this nature were encountered during testing.

F. PROBLEM AREAS AND PROPOSED SOLUTIONS

Problems encountered with the $\text{OF}_2/\text{B}_2\text{H}_6$ MTI tests center mainly around the tank injector hardware. These include injector port plugging due to freezing and long couple distance from the start valve to the injector port that produces long start transient and poor flow control. Subscale glassware tests were conducted to supply information about the reaction process, flame speed, and ignition delay with the propellants in liquid phase. This information provides the basis for the injector design to be used in future MTI pressurization/expulsion tests.

The injector design shown in Figure 28 is a very close-coupled system, allowing only a minimum amount of liquid to be trapped between the valve seat and the tank. The injector pressure drop must be high enough to assure that the stream velocity is above the flame velocity and/or provides a satisfactory tank entry distance during the ignition delay time. This procedure will assure a cool injector face so injector burnout is not a problem. The pressure drop large enough to achieve this effect may produce a flow rate too high for practical hole sizes. Should this problem develop, the weight flow rate could be reduced while maintaining flow velocity. One method to achieve this is to pulse the system (which will reduce the average flow rate). A second method is to inject a gas that (for the same stream size and velocity) provides a lower weight flow.

A separate problem that exists with all MTI injector designs is the start transient; here it is desirable to have the initial reaction occur at a point in the tank where there is no metal contact with the reaction. The injector in Figure 28 is designed to accomplish this as follows:

1. The trapped volume of the injector (from poppet seat to nozzle exit, assuming a $1/8$ in. length and 0.015 in. dia) is 36×10^{-5} cc.

2. The desired flow rate of OF_2 for a 10-second, 1.27-gallon expulsion of B_2H_6 is $16.8 \times 10^{-5} \text{ cc/millisecond}$ ($59.4 \times 10^{-5} \text{ lb/sec}$). This flow is achieved using a 0.012 in. dia orifice and a 100 psi pressure drop.

3. The trapped volume of $36 \times 10^{-5} \text{ cc}$ must be filled with OF_2 (and emptied of any B_2H_6 that has entered the port from the tank due to the low surface tension of the B_2H_6) in less time than the ignition delay of the $\text{OF}_2/\text{B}_2\text{H}_6$ reaction.

4. The glassware tests to date indicate a minimum ignition delay of one millisecond.

5. To inject at least twice the trapped volume in 1 millisecond ($36 \times 10^{-5} \text{ cc}$), the desired flow rate should be increased by a factor of 4.3 ($16.8 \times 10^{-5} \times 4.3 = 72 \times 10^{-6} \text{ cc/millisecond}$). This can be obtained by increasing the 100 psi injector pressure drop to approximately 430 psi.

6. The flow rate is controlled to the desired average value by pulsing the valve with a 1/4.3 or 23.3% duty cycle. This value allows flow during 23.3 millisecond out of every 100 millisecond.

The injector pulsing is required to maintain the 10-second expulsion time with 1.27 gallons of B_2H_6 and a 0.015 in. dia valve port; while it is felt that the valve port of 0.015 in. dia is a minimum, the pulsing would not be required if the expulsion total-volume-to-total-time ratio (1.27 gallons/10 seconds) were increased by a factor of at least 4.3 to one. Thus, the minimum expulsion volume flow (from the injector) for OF_2 into B_2H_6 (without pulsing) would be about 0.55 gallons/second of B_2H_6 .

Figure 29 presents the test setup schematic that would be used with this injector. A nitrogen cooled Freon-12 bath is utilized to maintain both propellants in a liquid state without incurring freezing problems.

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3. Analytical and Experimental Investigation of a Novel Liquid Propellant Pumping Concept, Aerojet-General Report No. 7409-01F, September 1962 (AFFTC Report SSD-TRD-62-76) (Confidential)
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9. Development and Demonstration of Main Tank Injection (MTI) Pressurization System, Contract AF 04(611)-8198, Report No. RTD-TDR-63-1123, December 1963 - Martin Company Denver Division

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TABLE 1

IMPULSE REQUIREMENTS

<u>Type of Maneuver or Function</u>	<u>Nominal Total Impulse 10⁶ lb-sec</u>	<u>Number of Starts</u>	<u>Storage Time days</u>
Lunar-Exploration Mission - Thrust Level, 10,000 to 60,000 lb			
Outbound trajectory correction	0.60	2-3	3-4
Outbound orbit injection	13.5	1	3-4
Perilunar variation	0.90	1	3-4
Landing from orbit	17.2	1	3-5
Hovering and transverse maneuvering	3.4	1	3-5
Lunar takeoff	11.3	1	3-10 (surface)
Return trajectory corrections	0.18	2-3	3-8
2.8% ullage and outage	1.32	-	3-10 (surface)
	<u>48.4</u>	-	-
Mars Orbital Mission - Thrust Level, 10,000 lb			
Outbound trajectory corrections	Small	-	-
Mars-orbit injection (including ullage and outage)	4.15	-	200
Total	<u>4.15</u>	-	-

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TABLE 2

THEORETICAL MISSION PERFORMANCE

	<u>LF₂/LH₂</u>	<u>OF₂/B₂H₆</u>
Chamber pressure (P _c), psia	100	200
Mixture ratio, (MR)	11.0	3.0
Specific impulse (I _{sp}), lb-sec/lb	477	439
Pressure ratio (P _c /P _e)	500	500
Area ratio (A _t /A _e)	50	50
Propellant-tank pressure, psia	200	300
Thrust level, lb	60K-10K, variable	10K, fixed

Table 2

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TABLE 3

MARS MISSION, WEIGHT COMPARISON

No.	Component Combination	Weight, lb
	Description	
13	Jet pump	452
10	Dual bipropellant gas generators	782
11	Solid-propellant gas generator	788
12	Solid-propellant gas generator (Bladder)	816
6	Main-tank injection (sequential)	869
8	Stored gas, (heated)	902
7	Main tank injection (connected ullages)	917
16	Stored gas (solid-propellant charge heated)	978
15	Stored gas (heated) and evaporated fuel	1082
2	Stored gas, blowdown	1135
3	Stored gas, blowdown	1136
1	Stored gas, regulated	1140
14	Stored gas (heated) and fuel-rich bipropellant gas generator	1212
5	Saturated vapor (secondary Vapak)	5048
9	Electromechanically actuated bellows tanks	9480
4	Saturated vapor, (Vapak)	*

* Combination 4 could not meet the required pressure; its weight was therefore not calculated.

Table 3

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TABLE 4

LUNAR MISSION, WEIGHT COMPARISON

Component Combination		
No.	Description	Weight, lb
13	Jet pump	6,792
10	Dual bipropellant gas generator	9,486
7	Main tank injection (connected ullages)	9,881
6	Main tank injection (sequential)	10,750
14	Stored gas (heated) and fuel-rich bipropellant gas generator	14,417
16	Stored gas (solid-propellant-heated)	15,032
8	Stored gas, (heated)	15,305
15	Stored gas (heated) and evaporated fuel	16,371
2	Stored gas, blowdown	17,045
3	Stored gas, blowdown	17,046
1	Stored gas, regulated	17,050
9	Electromechanically actuated bellows tanks	18,300
11	Solid-propellant gas generator	20,000
12	Solid-propellant gas generator (bladder)	20,000
5	Saturated vapor (secondary Vapak)	50,983
4	Saturated vapor (Vapak)	*

* Combination 4 could not meet the required pressure; its weight was therefore not calculated.

Table 4

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TABLE 5

MARS MISSION, VOLUME COMPARISON

Component *	Volume, cu ft					
	CC 10 **	CC 6	CC 12	CC 11	CC 8	CC 16
Gas bottle	0.1	0.3	-	-	5.1	23.7
Auxiliary tank, fuel	0.9	-	-	-	-	-
Auxiliary tank, oxidizer	0.7	0.3	-	-	-	-
Propellant tank, fuel	87.0	87.0	87.2	87.0	87.0	87.0
Propellant tank, oxidizer	68.2	68.2	68.4	68.2	68.2	68.2
Gas generator, solid	-	-	1.0	1.0	-	-
Total	156.9	155.8	156.6	156.2	160.3	178.9

TABLE 6

LUNAR MISSION, VOLUME COMPARISON

Component *	Volume, cu ft					
	CC 10 **	CC 6	CC 14	CC 16	CC 8	CC 15
Gas bottle	33	4	29	386	26	27
Auxiliary tank, fuel	102	-	4	-	-	7
Auxiliary tank, oxidizer	38	8	31	-	-	-
Propellant tank, fuel	1940	1940	1940	1940	1940	1940
Propellant tank, oxidizer	1053	1053	1053	1053	1053	1053
Total	3166	3005	3057	3379	3019	3027

* The volumes of the other components were not considered because they are small in comparison with those considered here.

** CC 10 represents Component Combination 10.

Tables 5 and 6

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TABLE 7

MARS MISSION, COST COMPARISON

Component	** Note	CC 10*		CC 6		CC 12		CC 11		CC 8		CC 16	
		Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$
Disconnect	(1)	1	820	2	1,640	-	-	-	-	1	820	1	820
Gas bottle	(1)	1	1,800	1	500	-	-	-	-	1	3,900	1	3,900
Pressure regulator	(1)	1	10,000	1	10,000	-	-	-	-	1	10,000	1	10,000
Check valve	(1)	2	320	-	-	2	320	2	320	2	320	2	320
Auxiliary fuel tank	(1)	1	6,400	-	-	-	-	-	-	-	-	-	-
Auxiliary oxidizer tank	(1)	1	5,850	1	500	-	-	-	-	-	-	-	-
Solenoid valve	(2)	4	2,000	2	1,000	-	-	-	-	1	500	1	500
Relief valve	(2)	2	400	2	400	1	200	1	200	2	400	2	200
Oxidizer tank	(1)	1	800	1	800	1	800	1	800	1	800	1	800
Fuel tank	(1)	1	1,000	1	1,000	1	1,000	1	1,000	1	1,000	1	1,000
Liquid gas generator	(2)	2	540	-	-	-	-	-	-	-	-	-	-
Injector head	(1)	-	-	2	250	-	-	-	-	-	-	-	-
T.C. heat exchanger	(2)	-	-	-	-	-	-	-	-	1	1,300	-	-
Igniter W/SA	(2)	-	-	-	-	1	600	1	600	-	-	1	600
Solid grain	(3)	-	-	-	-	-	-	-	-	-	-	1	250
Switch	(2)	-	-	-	-	-	-	-	-	-	-	1	400
Solid gas generator	(3)	-	-	-	-	1	350	1	350	-	-	-	-
Filter	(1)	-	-	-	-	1	1,000	1	1,000	-	-	-	-

* CC 10 represents Component Combination 10.

** Cost references in this column are defined at the end of this table.

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TABLE 7 (cont.)

Component	Note *	CC 10		CC 6		CC 12		CC 11		CC 8		CC 16	
		Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$
Burst diaphragm	(2)	-	-	-	-	1	15	1	15	-	-	-	-
Orifice	(2)	-	-	-	-	1	70	1	70	-	-	-	-
Bladder	(2)	-	-	-	-	2	15,000	-	-	-	-	-	-
Total			29,930		16,090		19,355		4,355		19,040		18,790

* Cost References

- (1) Miscellaneous vendor quotations on the same and similar hardware.
- (2) Design Guide for Pressurization System Evaluation, Liquid-Propellant-Fed Rocket Engines, Aerojet-General Corporation Report 2334, Volume III, 30 September 1962 (Confidential).
- (3) Engineering estimate.

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TABLE 8

LUNAR MISSION, COST COMPARISON

Component	** Note	CC 10*		CC 6		CC 14		CC 16		CC 8		CC 15	
		Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$	Qty	Cost, \$
Disconnect	(1)	1	820	2	820	1	820	1	820	1	820	1	820
Gas Bottle	(1)	1	5,000	1	2,500	1	4,200	1	11,000	1	4,500	1	3,100
Pressure regulator	(1)	1	10,000	1	10,000	1	10,000	1	10,000	1	10,000	1	10,000
Check valve	(1)	2	320	1	160	2	320	2	320	2	320	2	320
Auxiliary fuel tank	(1)	1	4,000	-	-	1	4,800	-	-	-	-	1	4,200
Auxiliary oxidizer tank	(1)	1	4,000	1	2,500	1	2,400	-	-	-	-	-	-
Solenoid valve	(2)	4	2,000	2	1,000	2	1,000	1	500	1	500	1	500
Relief valve	(2)	2	400	2	400	2	400	2	400	2	400	2	400
Oxidizer tank	(1)	1	15,100	1	15,100	1	15,100	1	15,100	1	15,100	1	15,100
Fuel tank	(1)	1	18,500	1	18,500	1	18,500	1	18,500	1	18,500	1	18,500
Liquid gas generator	(2)	2	440	-	-	1	270	-	-	-	-	-	-
Injector head	(1)	-	-	2	250	-	-	-	-	-	-	-	-
T.C. heat exchanger	(2)	-	-	-	-	-	-	-	-	1	1,300	2	2,200
Igniter W/SA	(2)	-	-	-	-	-	-	1	600	-	-	-	-
Solid grain	(3)	-	-	-	-	-	-	1	500	-	-	-	-
Switch	(2)	-	-	-	-	-	-	1	400	-	-	-	-
Valve, three-way	(3)	-	-	-	-	-	-	-	-	-	-	1	800

* CC 10 represents Component Combination 10.

** Cost references in this column are defined at the end of this table.

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TABLE 8 (cont.)

Component	Note	Qty Cost, \$		Qty Cost, \$		Qty Cost, \$		Qty Cost, \$		Qty Cost, \$		Qty Cost, \$		Qty Cost, \$		Qty Cost, \$		Qty Cost, \$		Qty Cost, \$		Total
	*	CC 10		CC 6		CC 14		CC 16		CC 8		CC 15		CC 16		CC 8		CC 15		CC 15		
Orifice	(2)	-	-	-	-	1	70	-	-	-	-	-	-	-	-	-	-	-	-	-	-	
Heat exchanger	(2)	-	-	-	-	1	1,300	-	-	-	-	-	-	-	-	-	-	-	-	-	-	
		60,580		51,230		58,180		58,140		51,440		55,940		58,140		51,440		55,940		55,940		

* Cost Reference

- (1) Miscellaneous vendor quotations on the same and similar hardware.
 (2) Design Guide for Pressurization-System Evaluation, Liquid-Propellant-Fed Rocket Engines, Aerojet-General Report 2334, Volume III, 30 September 1962.
 (3) Engineering estimate.

TABLE 9
MARS MISSION - RELIABILITY COMPARISON

Component Name	COMB. 10			COMB. 6			COMB. 12			COMB. 11			COMB. 8			COMB. 16							
	Hours		Reliability			Reliability			Reliability			Reliability			Reliability								
	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast	Qty					
Disconnect	0.0	4800	1	--	.9800	2	--	(.9800) ²						1	--	.9800	1	--	.9800				
Gas Bottle	4800	0	1	.9300	--	1	.9300	--					1	.9300	--	1	.9300	--					
Pressure Regulator	.12	4799.88	1	.9998	.9860	1	.9998	.9860					1	.9998	.9860	1	.9998	.9860					
Check Valve	4800	0	2	(.3700) ²		1	.3700	--	2	(.3700) ²		--	2	(.3700) ²		--	2	(.3700) ²					
Aux. Fuel Tank	4800	0	1	(.9300)		--																	
Aux. Oxidizer Tank	4800	0	1	(.9300)		--	1	.9300	--														
Solenoid Valve	.12	4799.88	4	(.9993) ⁴		(.9999) ⁴		2	(.9993) ²		(.9999) ²			1	.9993	.9999	1	.9993	.9999				
Relief Valve	4x10 ⁻⁸	4800	2	(.9999) ²		(.5000) ²		2	(.9999) ²		(.5000) ²		1	.9999	.5000	1	.9999	.5000	2	(.9999) ²		(.5000) ²	
Oxidizer Tank	4800	0	1	.9300	--	1	.9300	--	1	.9300	--	1	.9300	--	1	.9300	--	1	.9300	--			
Fuel Tank	4800	0	1	.9300	--	1	.9300	--	1	.9300	--	1	.9300	--	1	.9300	--	1	.9300	--			
Liquid Gas Generator	.12	4799.88	2	(.9999) ²		(.9800) ²																	
Injector Head	.12	4799.88				2	(.9999) ²		(.9983) ²														
T.C. Heat Exchanger	.12	4799.88											1	.9999	.9960								
Igniter		4800							1	--	.9800	1	.9800				1	--	.9800				
Solid Grain	.05	4799.95															1	.9999	.9000				
Switch	4800	0															1	.6800	--				
Solid Gas Generator	.12	4799.88							1	.9600	.6000	1	.9600	.6000									
Filter	.12	4799.88							1	.9999	.9800	1	.9999	.9800									
Burst Diaphragm	4800	0							1	.9480	--	1	.9480	--									
Orifice	.12	4799.88							1	.9999	.9983	1	.9999	.9983									
Bladder	4800	0							2	(.5000) ²		--											
Reliability				.0944	.2366		.2758	.2357		.0268	.2876		.1074	.2876		.1097	.2405		.0745	.2129			
Reliability (Oper. + Coast)																							
Non-Redundant				.0223			.0650			.0077			.0309		.0264				.0159				

TABLE 10
LUNAR MISSION - RELIABILITY COMPARISON

Component Name	Hours		COMB. 10			COMB. 6			COMB. 14			COMB. 16			COMB. 8			COMB. 15		
			Reliability			Reliability			Reliability			Reliability			Reliability			Reliability		
	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast	Qty	Oper	Coast
Disconnect	0	336	1	--	.9986	2	--	(.9986) ²	1	--	.9986	1	--	.9986	1	--	.9986	1	--	.9986
Gas Bottle	336	0	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--
Pressure Regulator	.26	335.74	1	.9995	.8300	1	.9995	.8300	1	.9995	.8300	1	.9995	.8300	1	.9995	.8300	1	.9995	.8300
Check Valve	336	0	2	(.6000) ²	--	1	.6000	--	2	(.6000) ²	--	2	(.6000) ²	--	2	(.6000) ²	--	2	(.6000) ²	--
Aux. Fuel Tank	336	0	1	.9950	--				1	.9950	--							1	.9950	--
Aux. Oxi- dizer Tank	336	0	1	.9950	--	1	.9950	--	1	.9950	--									
Solenoid Valve	.26	335.74	4	(.9985) ⁴	(.9950) ⁴	2	(.9985) ²	(.9950) ²	2	(.9985) ²	(.9950) ²	1	.9985	.9950	1	.9985	.9950	1	(.9985)	.9950
Relief Valve	8x10 ⁻⁴	336	2	(.9999) ²	(.9500) ²	2	(.9999) ²	(.9500) ²	2	(.9999) ²	(.9500) ²	2	(.9999) ²	(.9500) ²	2	(.9999) ²	(.9500) ²	2	(.9999) ²	(.9500) ²
Oxidizer Tank	336	0	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--
Fuel Tank	336	0	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--	1	.9950	--
Liquid Gas Generator	.26	335.74	2	(.9999) ²	(.9999) ²				1	.9999	.9999									
Injector Head	.26	335.74				2	(.9999) ²	.9990												
T.C. Heat Exchanger	.26	335.74													1	.9999	.9970	2	(.9999) ²	(.9970) ²
Igniter	0	336										1	--	.9985						
Solid Grain	.05	335.95										1	.9999	.8300						
Switch	336	0										1	.9991	--						
3-Way Valve	.26	335.74																1	.9985	.9950
Orifice	.26	335.74							1	.9999	.9988									
Heat Exchanger	.26	335.74							1	.9999	.9970									
Reliability				.5807	.7326		.5856	.7396		.3491	.7370		.2820	.6166		.3536	.7419		.3511	.7358
Reliability (Oper + Coast) Non-Redundant				.4254			.4331			.2573			.1739			.2623			.2583	

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TABLE 11

RATING-FACTOR SUMMARY

<u>Component Combination</u>	<u>Weight lb</u>	<u>Volume ft³</u>	<u>Reliability %</u>	<u>Cost \$</u>
Lunar Mission				
6	10,750	3005	0.4331	51,230
8	15,305	3019	0.2623	51,440
10	9,486	3166	0.4254	60,580
14	14,417	3057	0.2573	58,180
15	16,371	3030	0.2583	55,940
16	15,032	2879	0.1739	58,140
Mars Mission				
6	869	156	0.0650	16,090
8	902	160	0.0264	19,040
10	782	157	0.0223	29,930
11	788	156	0.0309	4,355
12	816	157	0.0077	19,355
16	978	179	0.0159	18,790

Table 11

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TABLE 12

INFLUENCE-COEFFICIENT SUMMARY

Component Combination	Coefficient				Final Influence Rating
	Weight	Volume	Reliability	Cost	
Lunar Mission					
6	6.7	1.25	11.4	1.60	153
10	7.8	1.10	11.1	1.55	148
14	3.3	1.20	4.7	1.30	24
8	2.5	1.24	4.9	1.55	23
15	1.3	1.22	4.75	1.35	10
16	2.8	1.40	1.6	1.30	8
Mars Mission					
6	2.9	1.95	11.1	1.6	100
11	4.1	1.95	4.8	2.0	77
10	4.2	1.90	3.2	1.0	26
8	2.5	1.80	4.0	1.4	25
16	1.3	1.30	2.1	1.4	5
12	3.7	1.90	0	1.4	0

Table 12

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TABLE 13

MTI IGNITION DELAY TEST FILM SUMMARY
(SUBSURFACE MTI, N_2O_4 INTO AEROZINE-50)

Frame No.	Time (millisec)	Action Recorded
0	0	No breakage of capillary tube or reaction evident
1	0.167	Blue-grey haze, 1/8 in. in diameter, at capillary bend; mixing, but no reaction
2	0.333	Blue-grey mixing haze enlarging
3	0.500	Same
4	0.667	Same
5	0.833	Possible localized blue reaction flame just starting near capillary break
6	1.000	Localized blue reaction flame or mixing haze enlarging
7	1.167	Same
8	1.333	Same
9	1.500	Same
10	1.667	Blue reaction flame, spreading downward
11	1.833	Definite blue reaction flame in bottom inch of tube
12	2.000	Same
13	2.167	Same; pressure surge forced N_2O_4 back down tube
14	2.333	Definite blue reaction flame in bottom inch of tube; N_2O_4 flow resumes and continues
15	2.500	Same
18	3.000	Same
36	6.00	Same
40	6.667	Reaction upper right appears to come from broken capillary tip
45	7.500	Same
46	7.667	Pressure surge forces N_2O_4 back down tube
46+		Fuel against walls, N_2O_4 stream across central ullage, hitting and reacting with fuel on far wall

TABLE 14

MTI IGNITION DELAY TEST FILM SUMMARY
(SUBSURFACE MTI, GAS-LIQUID OF F_2 INTO B_2H_6)

<u>Frame No.</u>	<u>Time (millisec)</u>	<u>Action Recorded</u>
0	0	No breakage of capillary tube or reaction evident
1	0.179	Definite reaction in hook of capillary tube
2	0.357	Hook reaction dying out; strong reaction moved to base of hook
3	0.536	Reaction at base of hook growing
4	0.714	Same
5	0.893	Test tube cracking
6	1.071	Test tube just broken and expanding
7	1.250	Test tube broken and expanding; reaction extending to left
8	1.429	Same
9	1.607	Interior of Dewar vessel beginning to break
10	1.786	Exterior of Dewar vessel beginning to break
11	1.964	All of Dewar vessel exterior breaking
12	2.143	Same
13	2.321	Same
14	2.450	Same; slight green coloring in or above reaction area
15	2.679	All of Dewar vessel exterior breaking

TABLE 15

MTI IGNITION DELAY TEST FILM SUMMARY
(SUBSURFACE MTI, GAS-LIQUID OF₂ INTO B₂H₆)

Split Frame No.	Time (millisec)	Action Recorded
0	0	No breakage of capillary tube or reaction evident
1	0.0833	Definite reaction in hook of capillary tube
2	0.167	Definite reaction in hook and extending down vertical section from hook
3	0.250	Hook reaction fading; reaction growing at base of vertical section
4	0.333	Same
5	0.417	Same
6	0.500	Reactions below-left of ball and left of small tube entrance into main tube; possible reaction near left of liquid at low point in small tube
7	0.583	
8	0.667	
9	0.750	Same
10	0.833	Same
11	0.917	
12	1.000	
13	1.083	Same, except reaction under ball widening outward
14	1.167	Ball moving upward; reaction extends from base of ball to liquid level near left of low point in small tube
15	1.250	
16	1.333	Ball rising rapidly; base of main tube has broken; liquid OF ₂ at low point of small tube reacting on left side
17	1.417	
18	1.500	Reaction brighter; main tube failure progressing from its base upward and down to low point of small tube
19	1.583	
20	1.667	Small tube broken down to low point
21	1.750	

TABLE 15 (cont.)

Split Frame No.	Time (millisec)	Action Recorded
22	1.833	Blur
23	1.917	
24	2.000	Interior of Dewar vessel beginning to break
25	2.083	
26	2.167	Interior of Dewar vessel shattered; exterior beginning to break
27	2.250	
28	2.333	Exterior of Dewar vessel breaking out on left front
29	2.417	
30	2.500	Exterior of Dewar vessel breaking all around
31	2.583	
32	2.667	Same
33	2.750	
34	2.833	Dewar vessel shattered and expanding mass; some green color
35	2.917	
36	3.000	Same
37	3.083	
38	3.167	Same
39	3.250	
40	3.333	Same
41	3.417	
42	3.500	Same; mass just reached only tape line on right
43	3.583	
44	3.667	Dewar vessel shattered and expanding mass
45	3.750	
46	3.833	Same; mass just reached first tape line on left
47	3.917	
48	4.000	Dewar vessel shattered and expanding mass

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TABLE 16

GAS GENERATOR IGNITION-DELAY-TEST RESULTS, OF $\text{F}_2/\text{B}_2\text{H}_6$

Run No.	Oxidizer		Fuel		O/F M.R.	Ignition Delay millisec	Pressure P, psig
	Injector Dia, in.	Tank Pressure psig	Injector Dia, in.	Tank Pressure psig			
1a	0.013	300	0.063	300	0.19	10	150
1b	0.013	300	0.063	300	0.19	8	100
1c	0.013	300	0.063	300	0.19	7	90
						Av 8	113
2a	0.063	200	0.013	400	17	0*	150*
2b	0.063	200	0.013	400	17	14	50
2c	0.063	200	0.013	400	17	7	40
						Av 11	45
3a	0.063	400	0.013	200	30	20	25
3b	0.063	400	0.013	200	30	24	25
3c	0.063	400	0.013	200	30	20	20
						Av 21	23
4a	0.063	800	0.013	200	40	12	40
4b	0.063	800	0.013	200	40	8	40
						Av 10	40

* Not included in average; may be in error due to air in chamber.

Table 16

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TABLE 17

ANALYSIS OF RESIDUE

(Test D-270-LQ-2)

	<u>Exit Tube %</u>	<u>Inside Tank %</u>
Fe	Major constituent	Major constituent
Al	0.03	0.04
Ni	0.05	0.06
Cu	0.002	0.003
Co	No	No
Cr	1.06	0.88
Si	0.10	0.07
Mg	0.008	0.015
Mn	0.5	0.5

Table 17

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TABLE 18

MTI ABOVE SURFACE B_2H_6 INTO OF_2

(Test D-270-LQ-2)

<u>Temperatures</u>		<u>FS-1 (°F)</u>	<u>FS-2 (°F)</u>	<u>FS-1 (°F)</u>	<u>Up to Blow (°F)</u>
Near Injector	T0-1	-	-	-	-
	T0-2	-129	-80	-98	-77
	T0-3	-120	-75	-95	-75
In OF_2 Tank	TOT-1	-276	-276	-276	-276
	TOT-2	-310	-310	-310	-310
	TOT-3	-	-	-	-
In Receiver Tank TRT-1		-322	-322	-322	-322
<u>Pressures</u>		<u>(psig)</u>	<u>(psig)</u>	<u>(psig)</u>	<u>(psig)</u>
B_2H_6 tank	P_{ft}	361	361	361	361
OF_2 tank	P_{otg}	311	311	311	311

INJECTOR LINE VOLUMES

AGC Bipropellant Valve (downstream from seat)	1.8 cc
Valve to Check Valve	0.6 cc
Check Valve	1.2 cc
Check Valve to Injector Orifice	<u>0.15 cc</u>
	3.75 cc

TABLE 19

 B_2H_6 FLOW (Test D-270-LQ-2)

<u>Recorded Flow</u>	Weight (10^{-4} lb)	Calculated Volume	
		at spgr = 0.42 (cc)	at spgr = 0.22 (cc)
1st Flow	5.04	0.544	1.039
2nd Flow	<u>3.96</u>	<u>0.428</u>	<u>0.817</u>
	9.0	0.972	1.856

Unrecorded Flow - Unobstructed Orifice

<u>Injected B_2H_6 Vapor Temperature (°F)</u>	<u>Unrecorded B_2H_6 Flow Rate (10^{-4} lb)</u>	<u>Total Unrecorded Flow (10^{-4} lb)</u>	<u>Ratio of Unrecorded to Recorded Flow</u>
30	0.844	30.4	3.38
100	0.789	28.4	3.18
200	0.732	26.4	2.93

Total Flow - Recorded and Unrecorded

<u>Injected B_2H_6 Vapor Temperature (°F)</u>	<u>Total B_2H_6 Flow (10^{-4} lb)</u>
30	39.4
100	37.4
200	35.4

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TABLE 20THEORETICAL REACTION PRESSURES FOR RECORDED PLUS MAXIMUM UNRECORDED B_2H_6 FLOW

	<u>M.R. = 3.0</u>	<u>M.R. = 120</u>
<u>Zero Mixing</u>		
P_{react}	990 psia	4970 psia
P_{He}	<u>326 psia</u>	<u>326 psia</u>
P_{T}	1316 psia	5296 psia
<u>Complete Mixing</u>		
P_{react}	14 psia	842 psia
P_{He}	<u>2400 psia</u>	<u>3200 psia</u>
P_{T}	2414 psia	4042 psia
T_{mix}	1300°R	1720°R

Table 20

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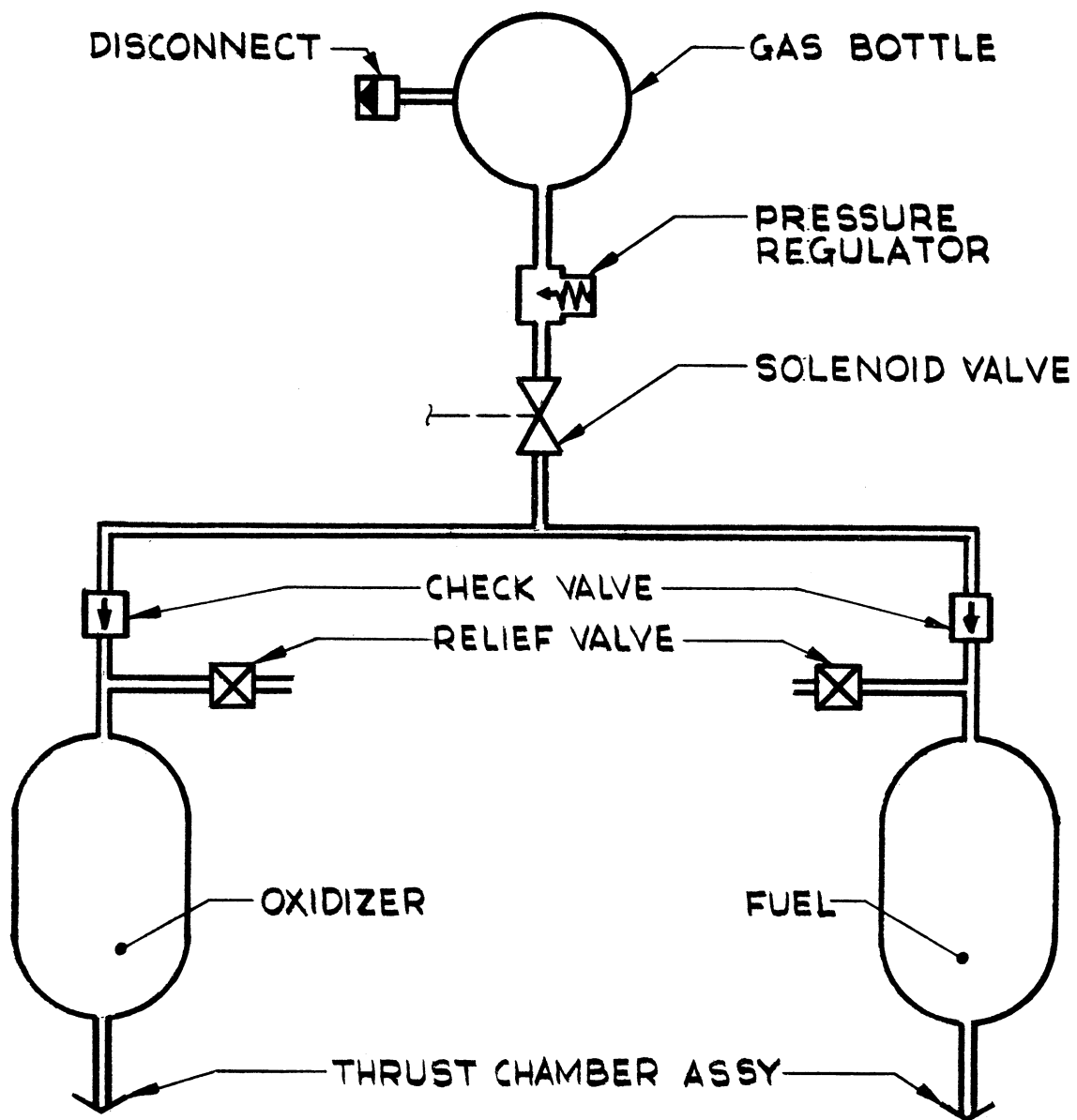
TABLE 21

COMPATIBILITY TEST SUMMARY

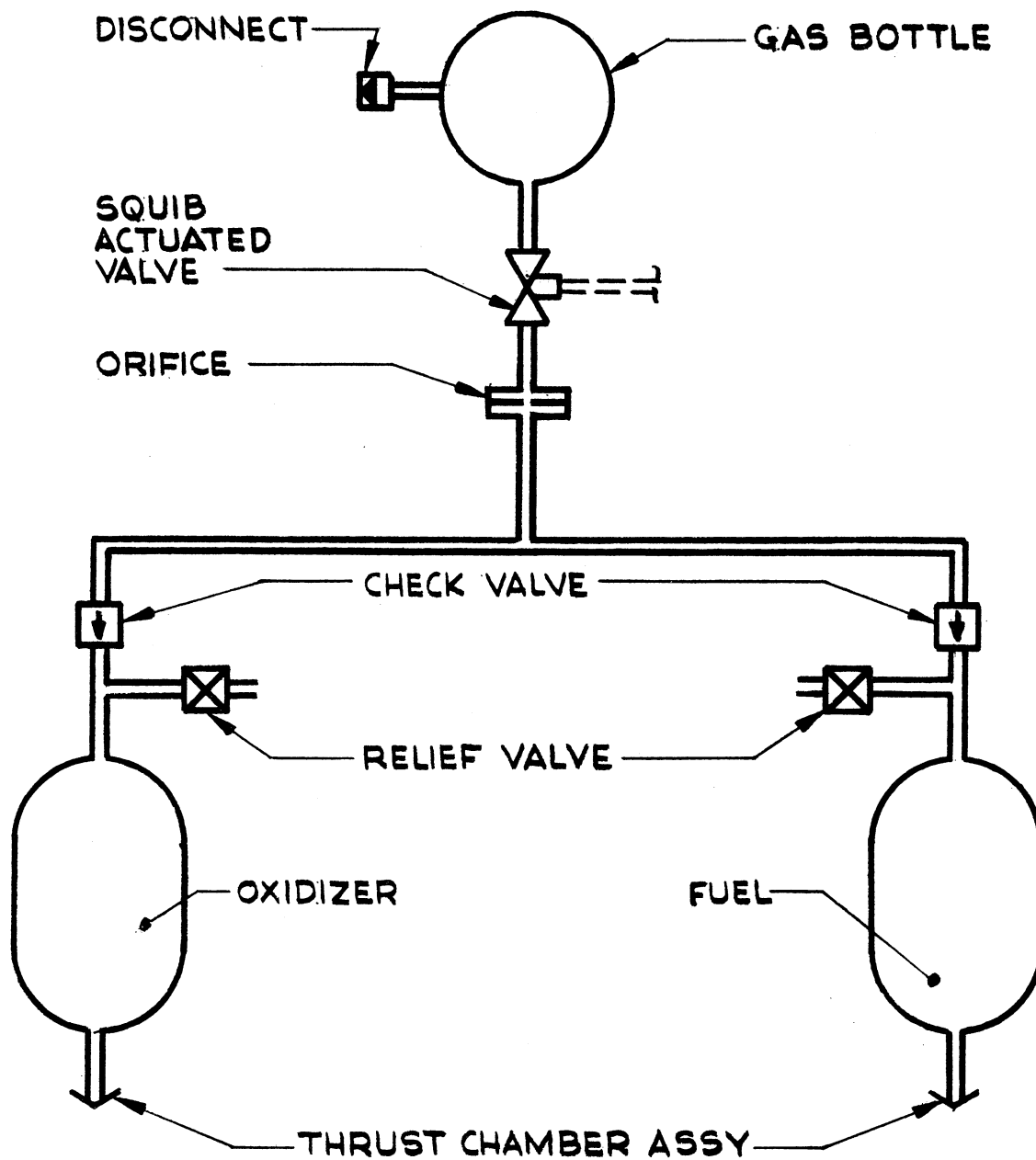
<u>Material</u>	<u>Gas Pressure, psi</u>	<u>Time, sec</u>	<u>Effect</u>
Scott Airpack	50	30	None
Gralite hood	80	30	None
Plexiglas face shield	50	30	None
Plexiglas face shield	80	30	None
Buna-N gloves	50	45	None
Plastic safety helmet	50	30	None
Kimwipes paper towel	50	5	None

Table 21

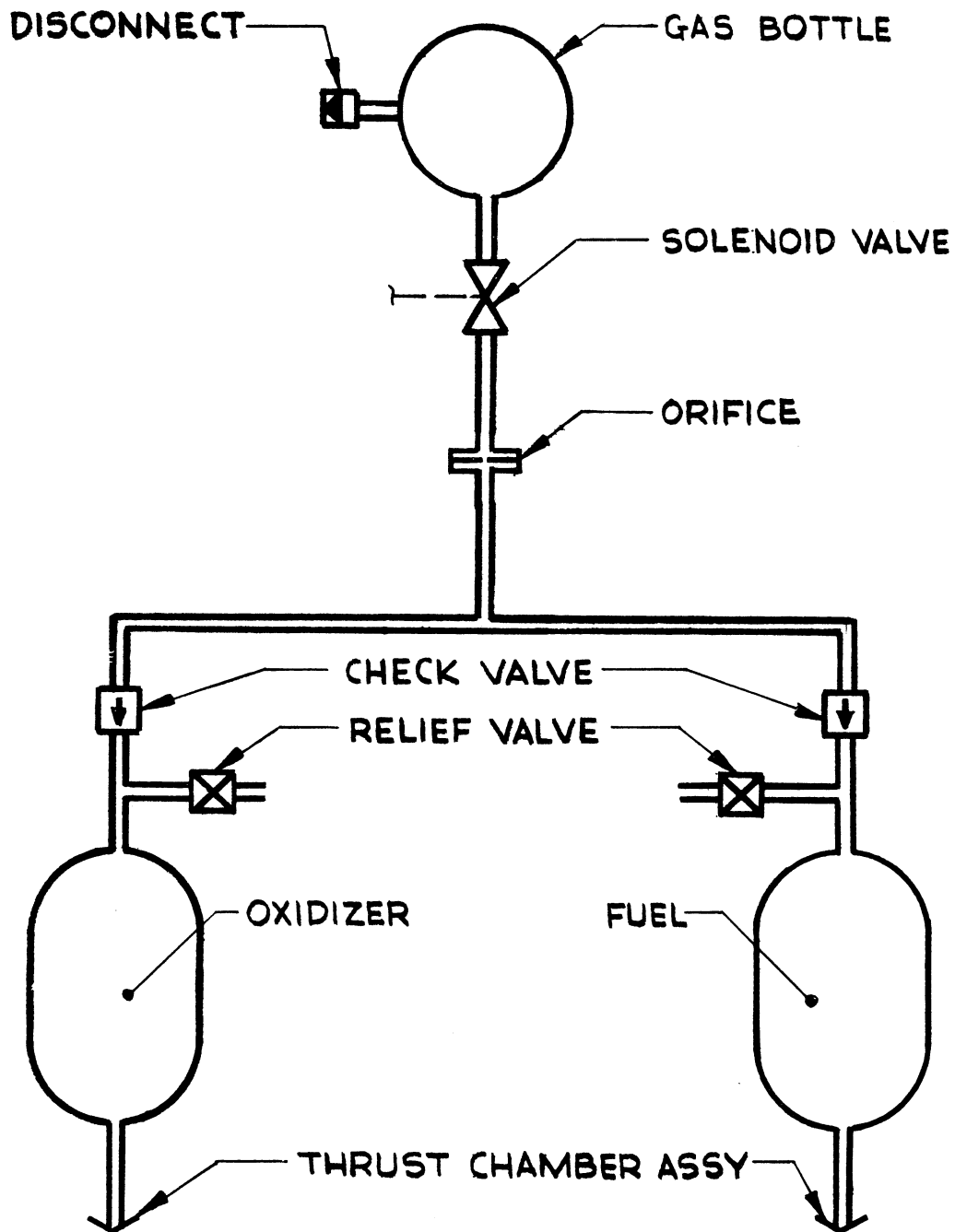
UNCLASSIFIED



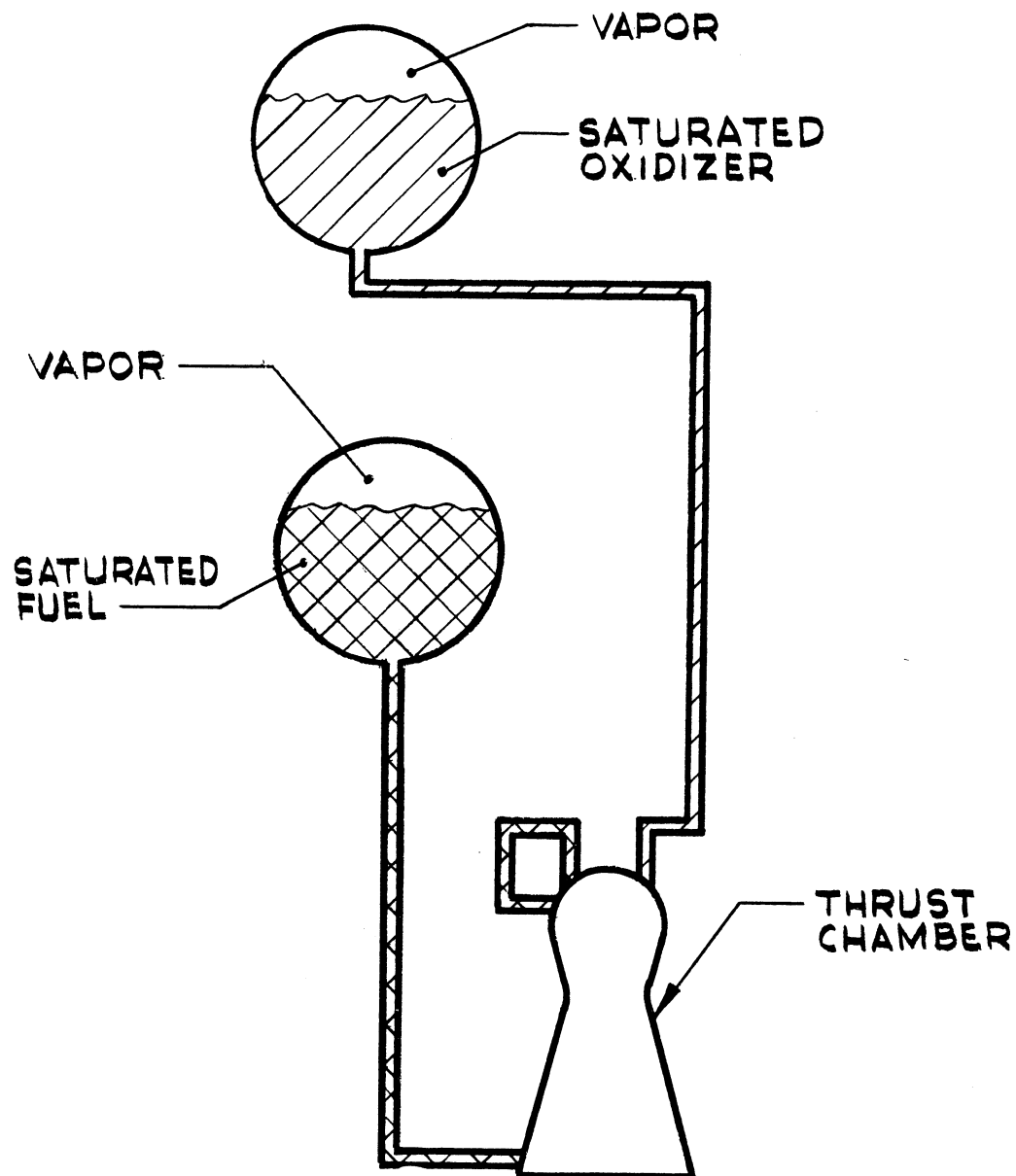
COMPONENT COMBINATION 1
STORED GAS, REGULATED



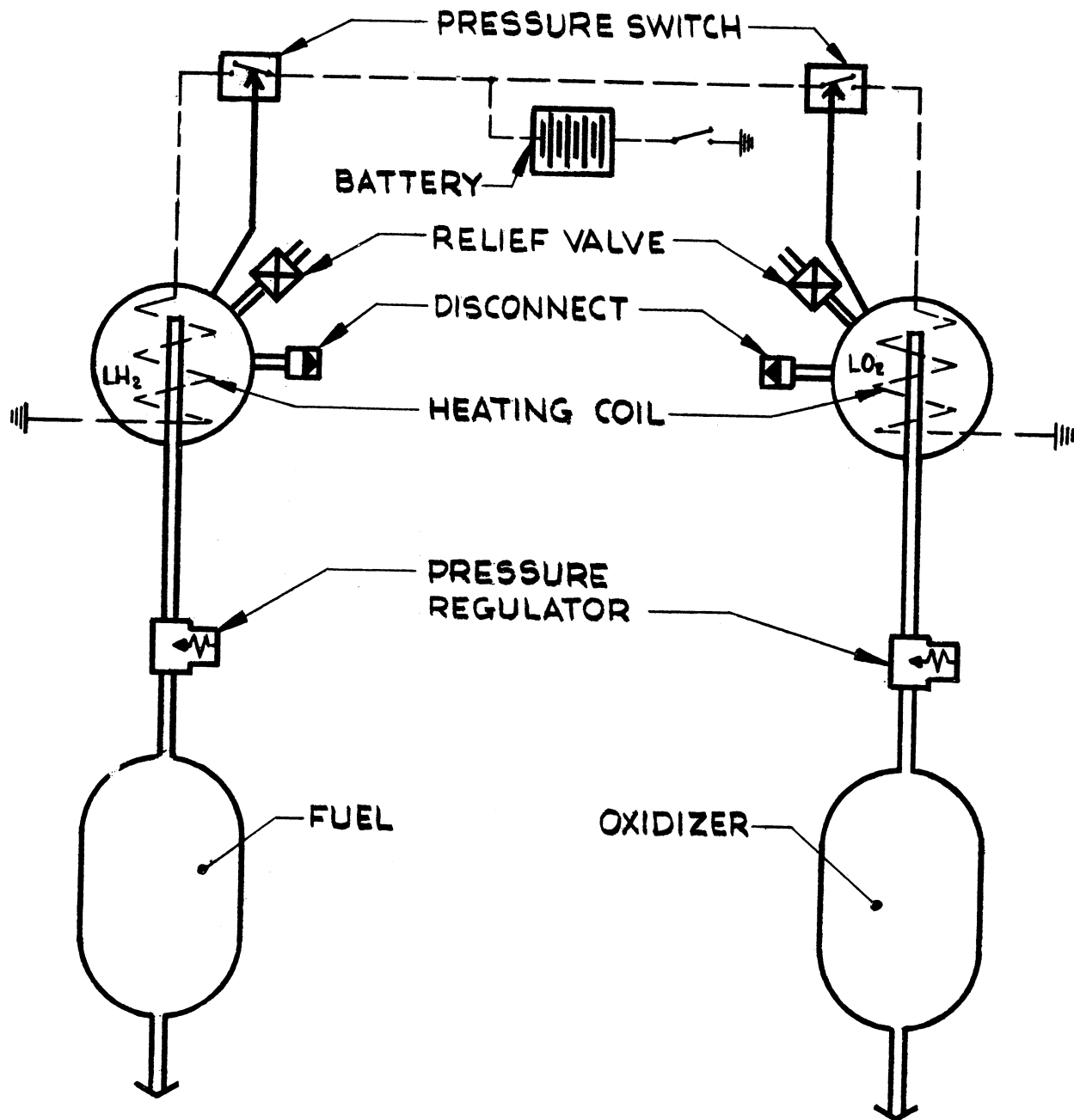
COMPONENT COMBINATION 2
STORED GAS, BLOWDOWN



COMPONENT COMBINATION 3
STORED GAS, BLOWDOWN



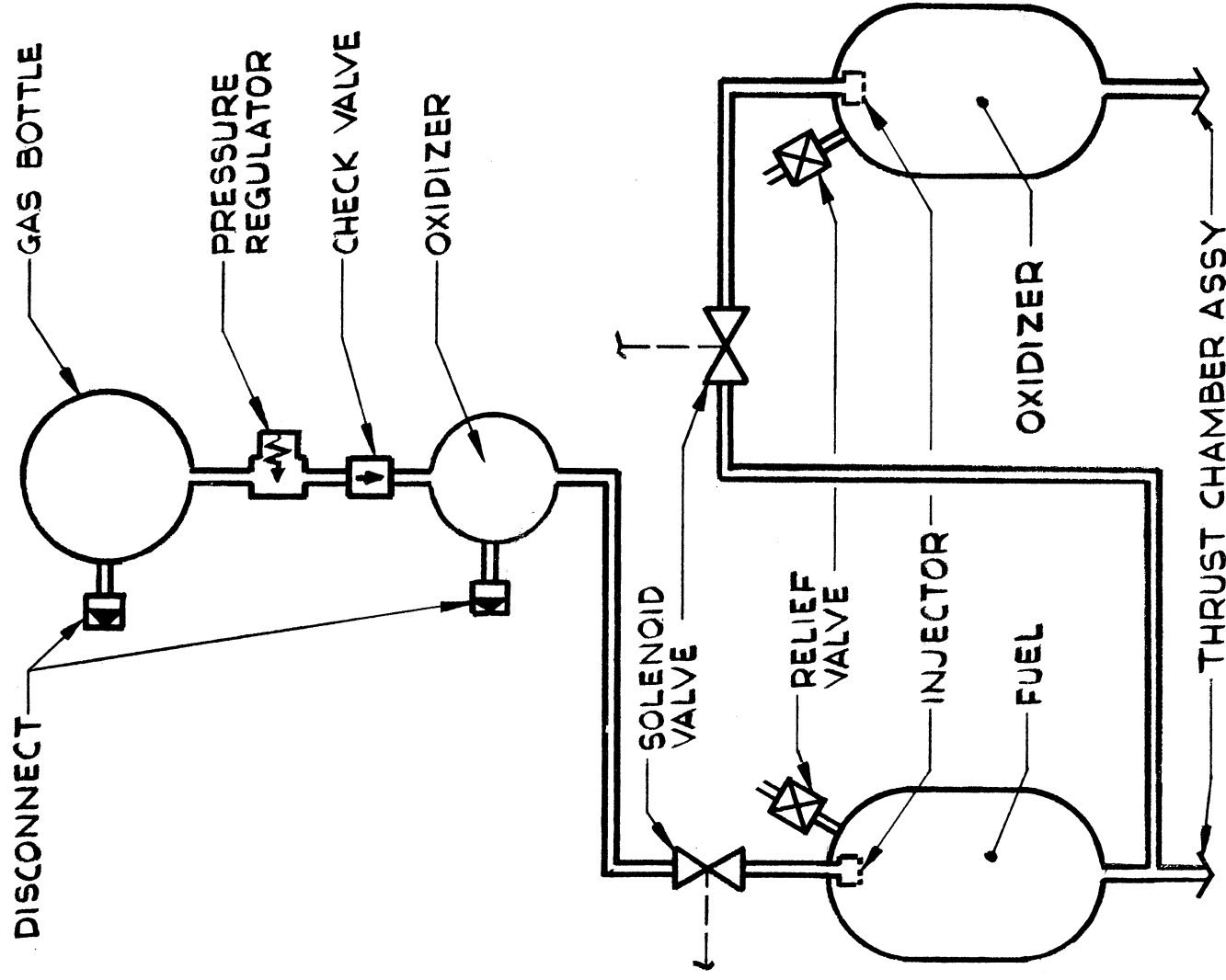
COMPONENT COMBINATION 4
SATURATED VAPOR (VĀPAK)



COMPONENT COMBINATION 5
SATURATED VAPOR (SECONDARY VAPOR)

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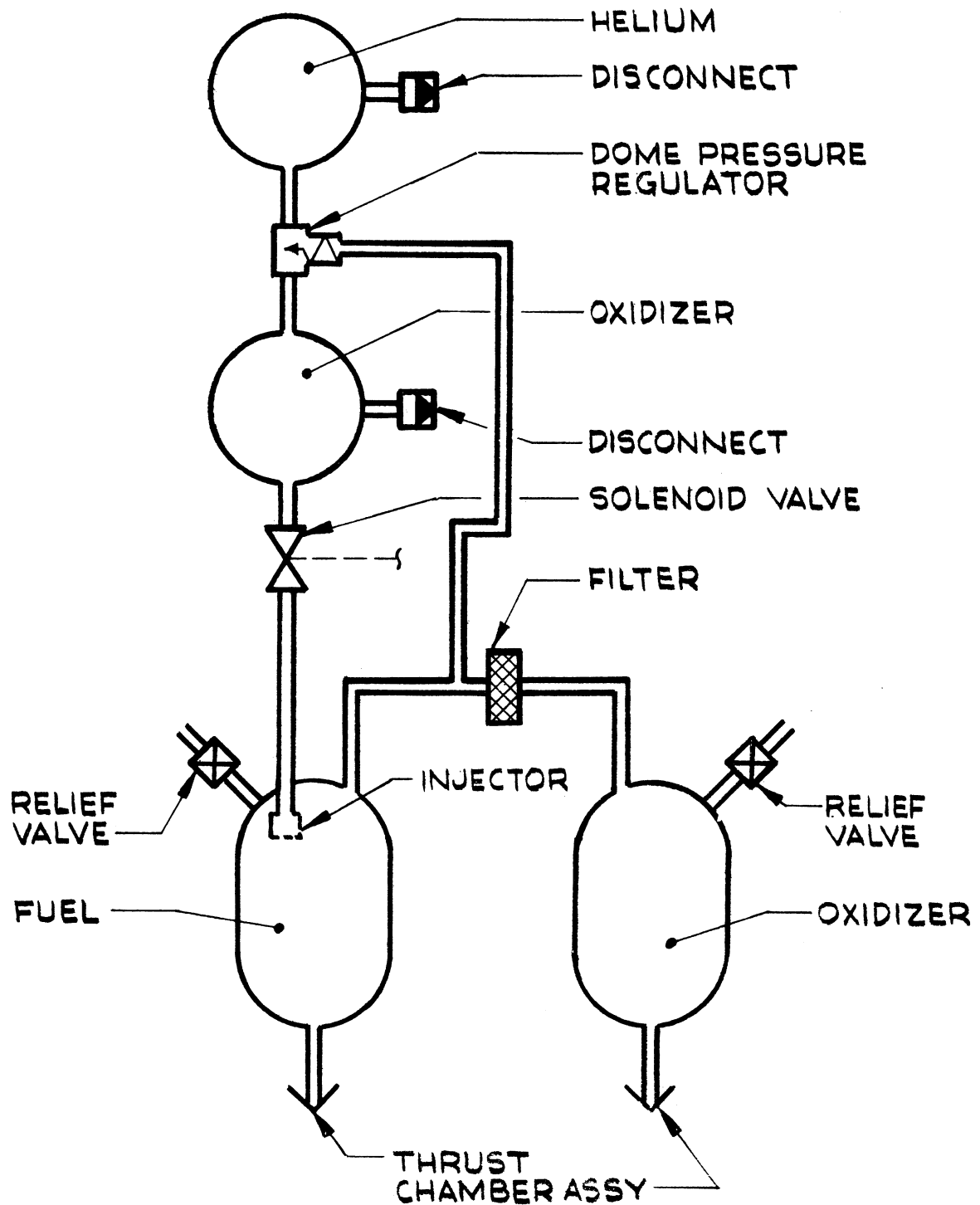
Report No. 2735



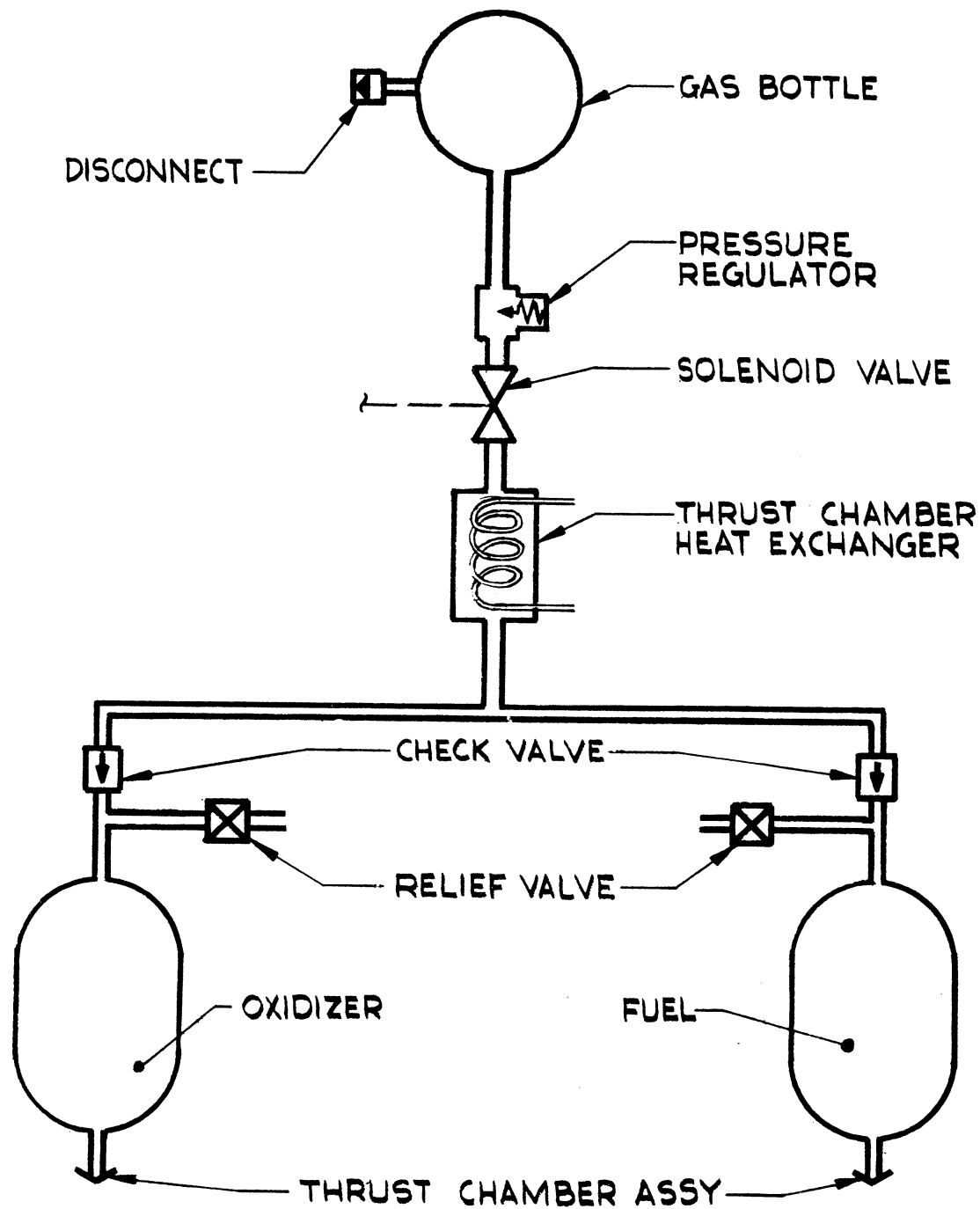
COMPONENT COMBINATION 6
MAIN TANK INJECTION(SEQUENTIAL)

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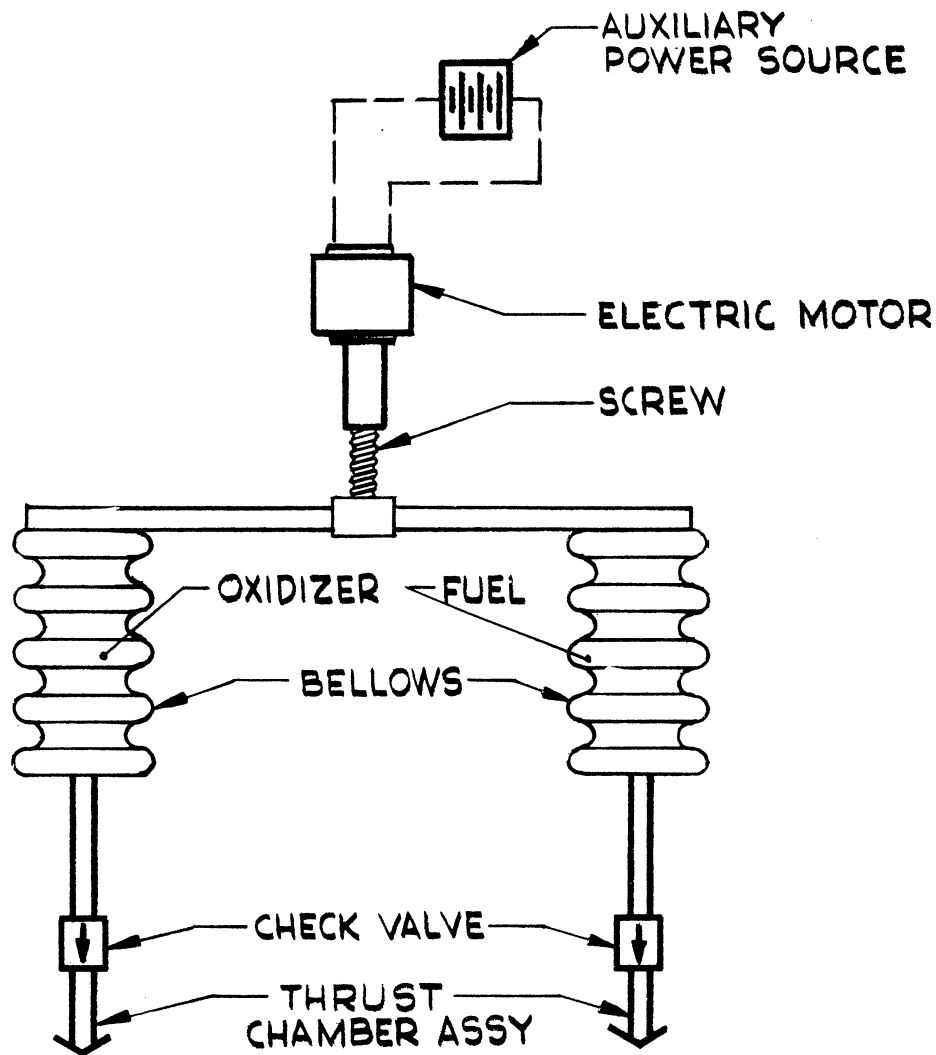
Figure 6



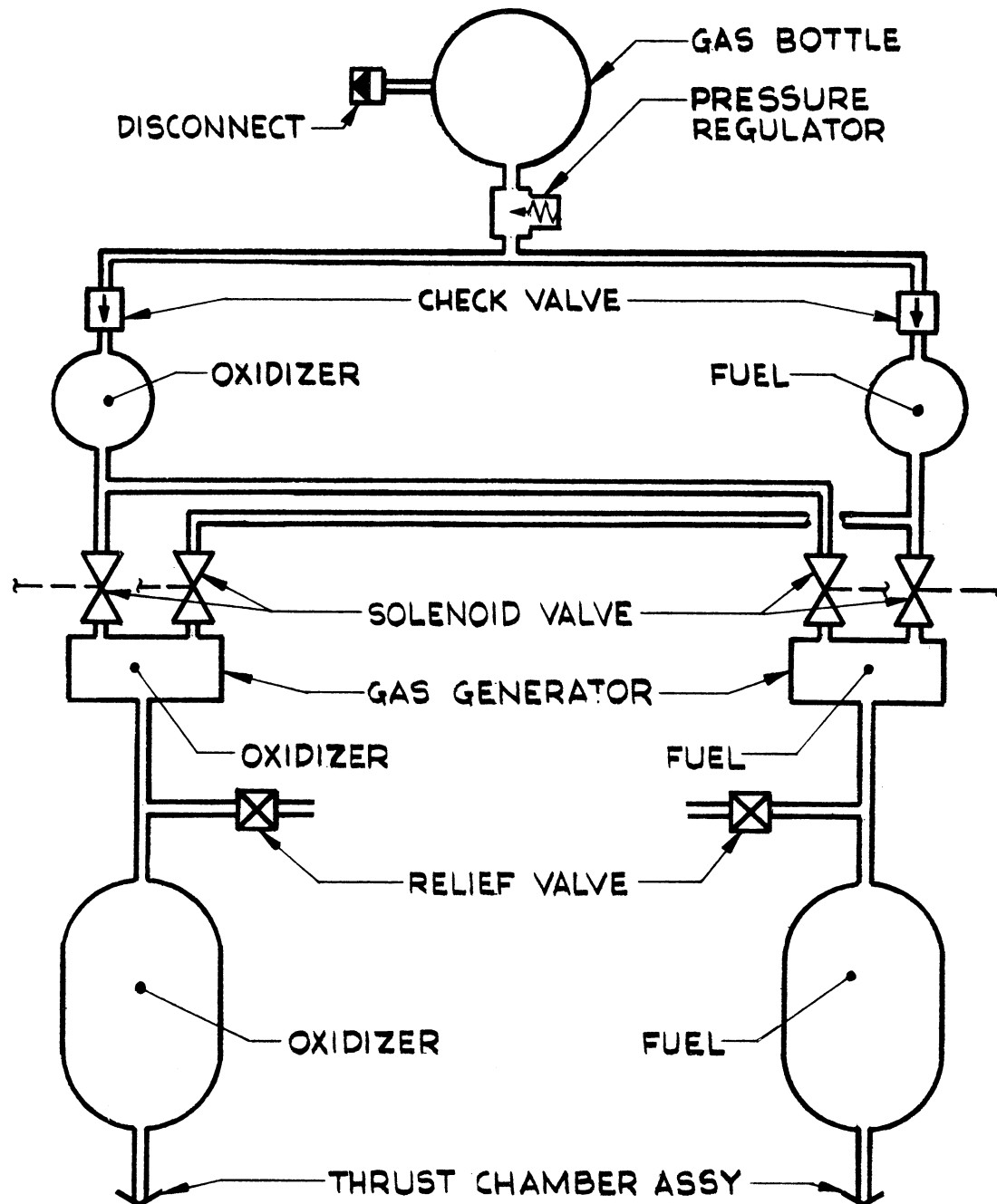
COMPONENT COMBINATION 7
MAIN TANK INJECTION(CONNECTED ULLAGES)



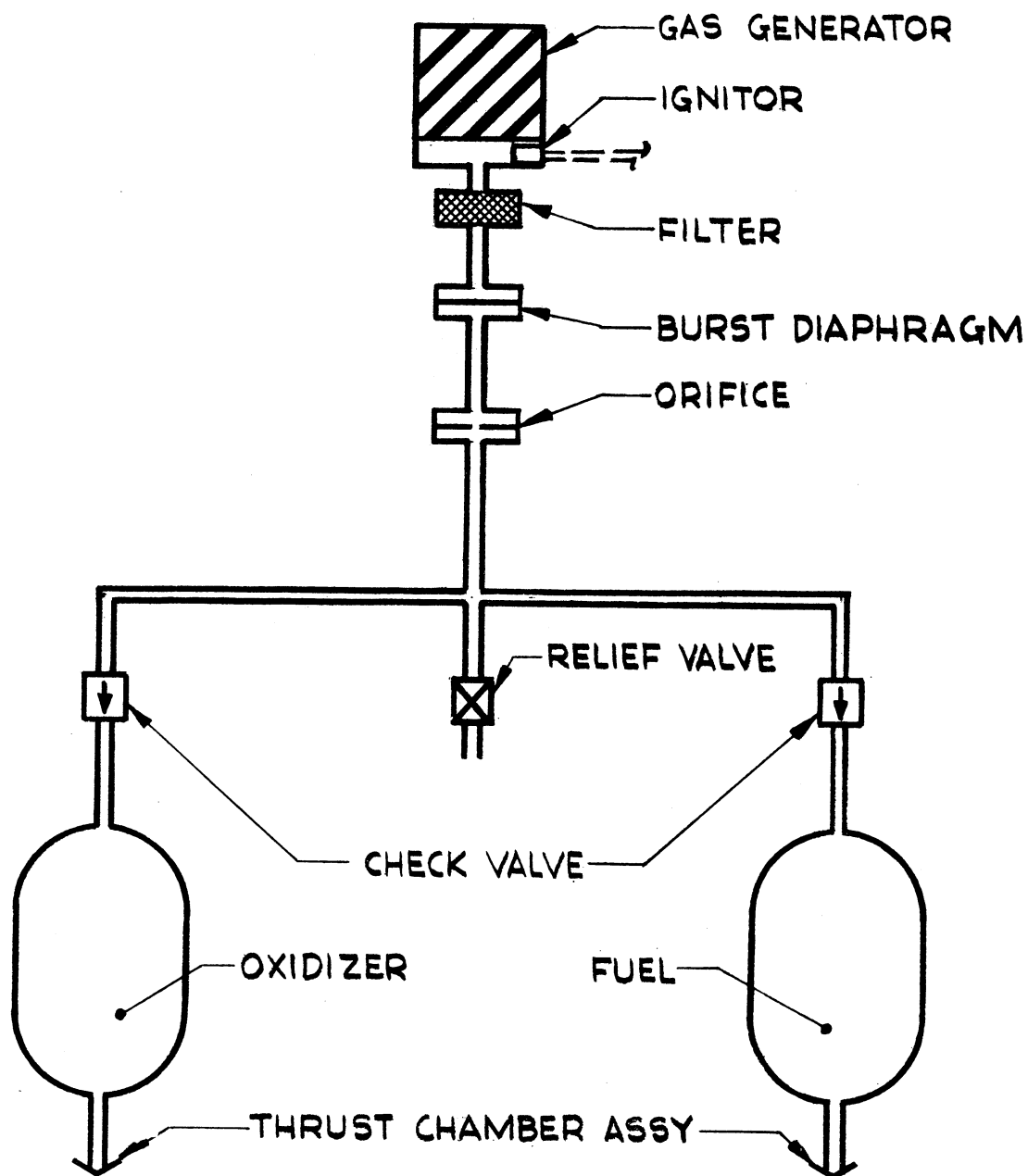
COMPONENT COMBINATION 8
STORED GAS (HEATED)



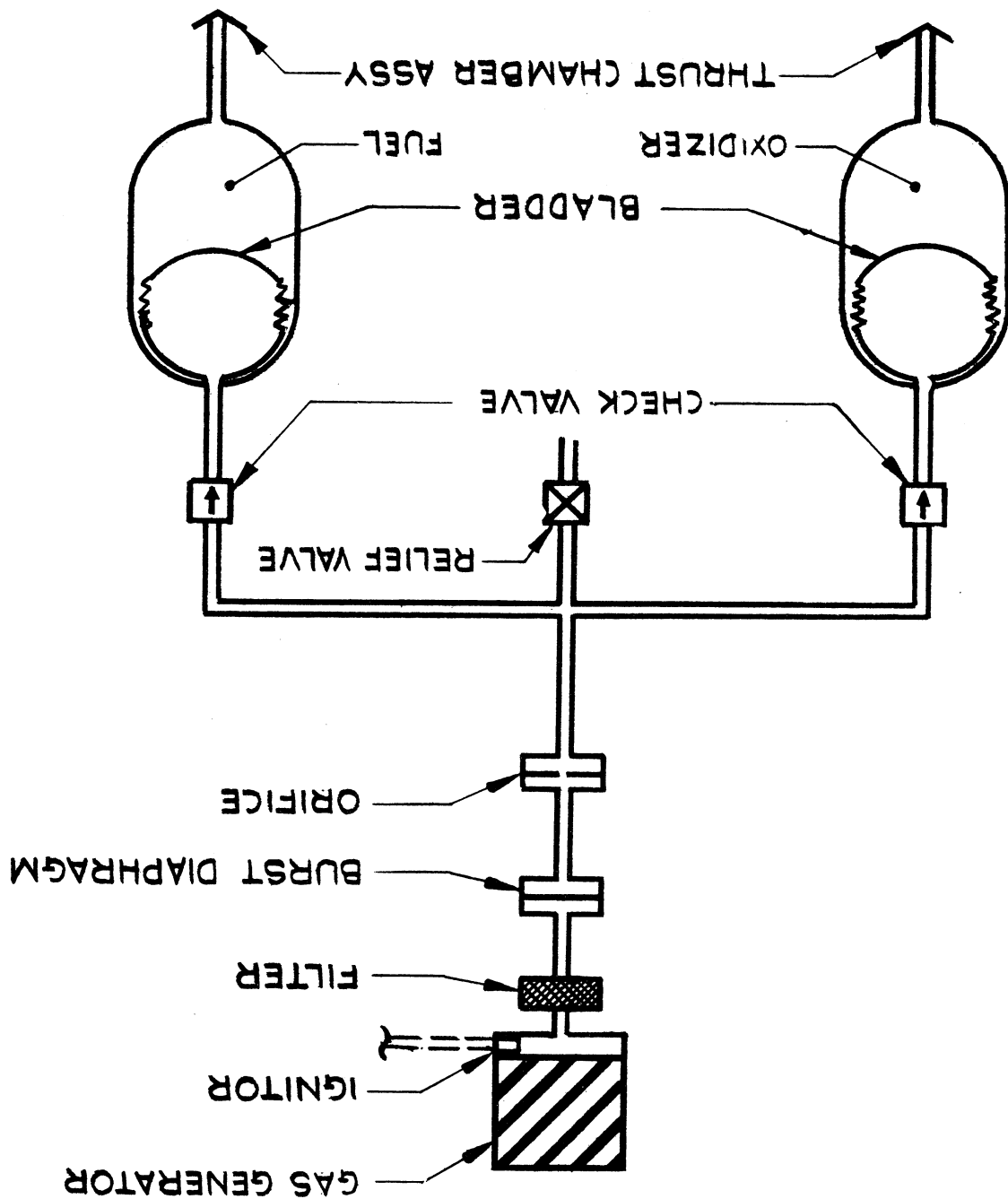
COMPONENT COMBINATION 9
ELECTRO-MECHANICALLY ACTUATED BELLOWS
TANKS



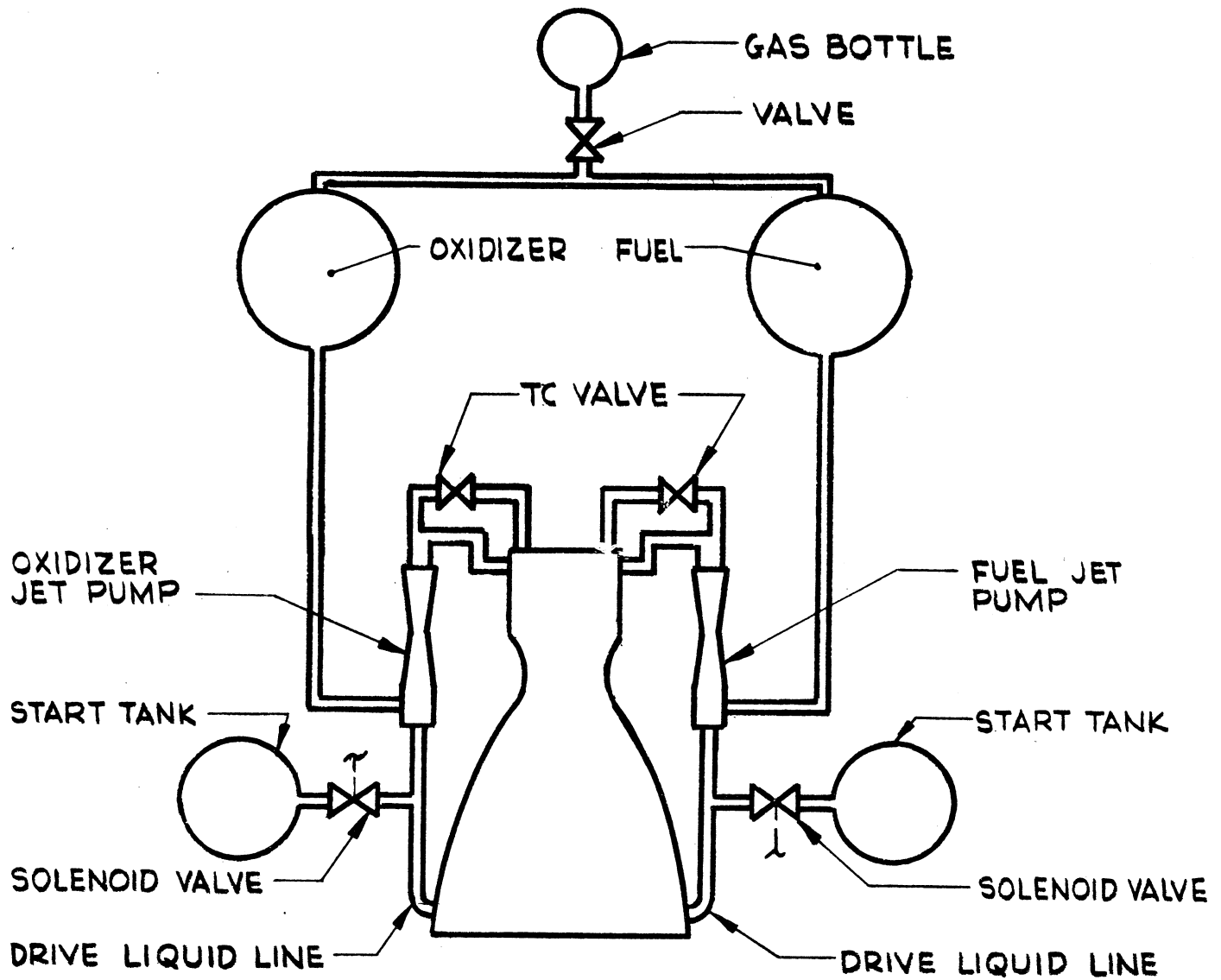
COMPONENT COMBINATION 10
DUAL BI-PROPELLANT GAS
GENERATORS



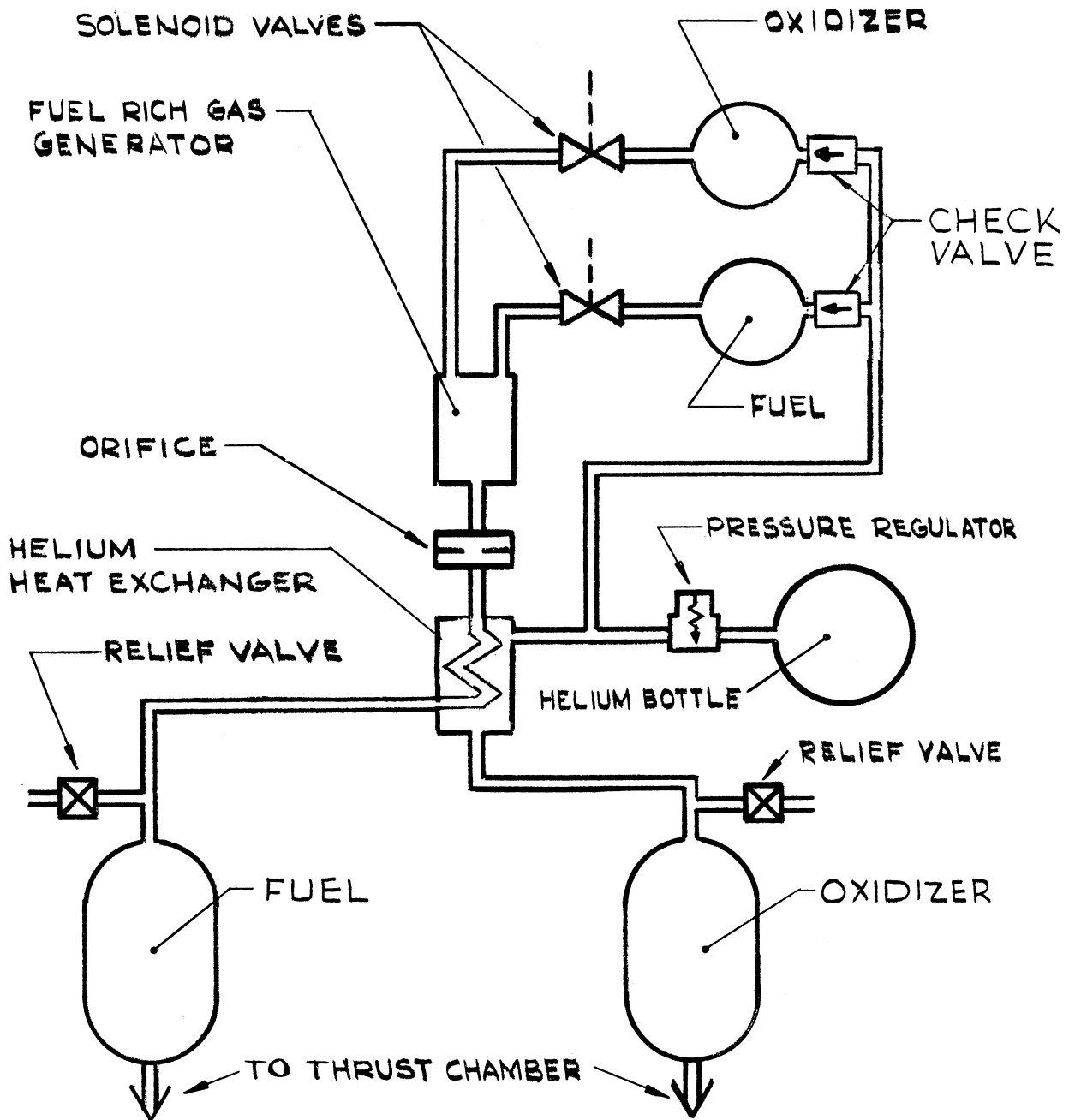
COMPONENT COMBINATION II
SOLID PROPELLANT GAS
GENERATOR



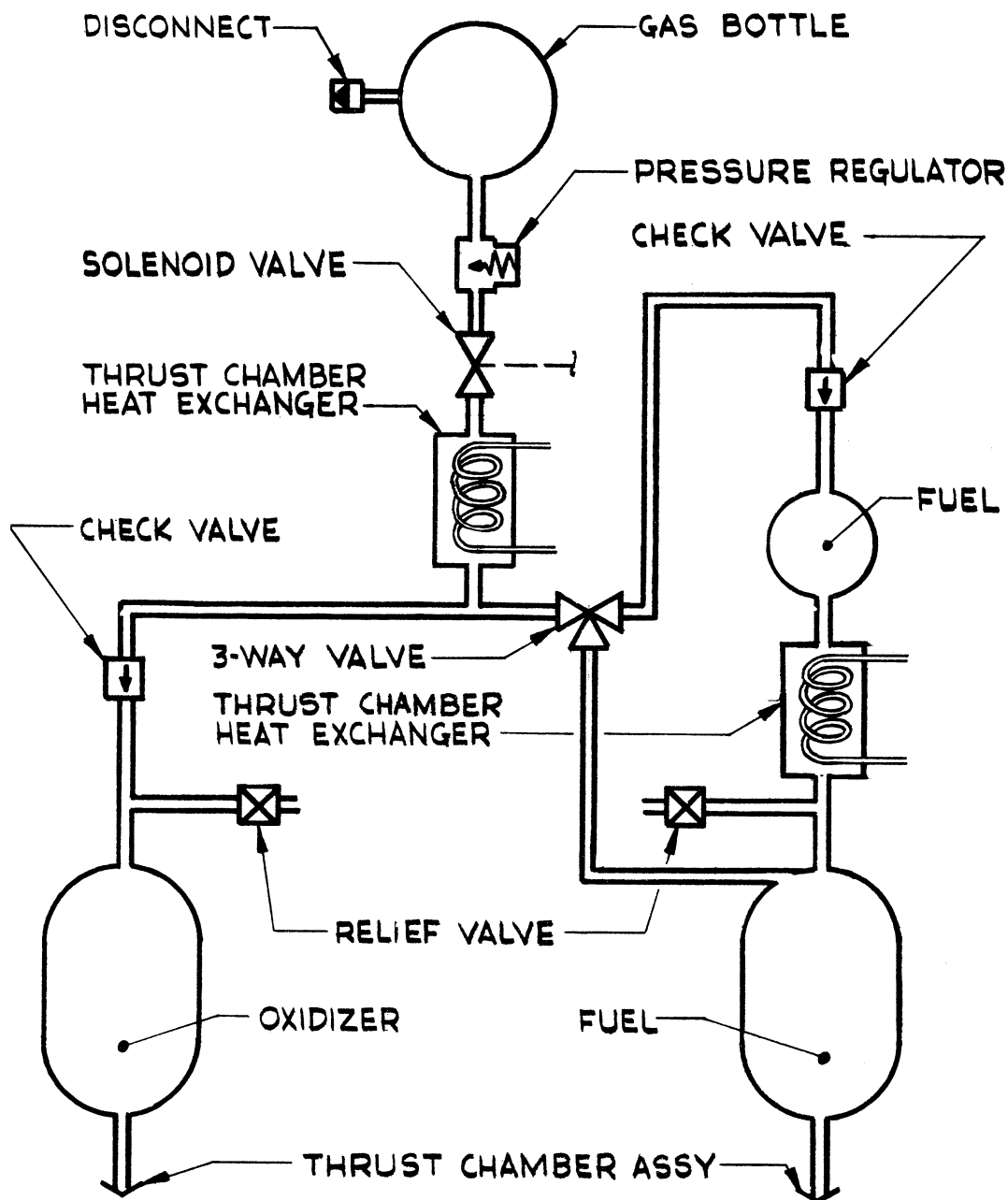
COMPONENT COMBINATION 12
SOLID PROPELLANT GAS
GENERATOR (BLADDER)



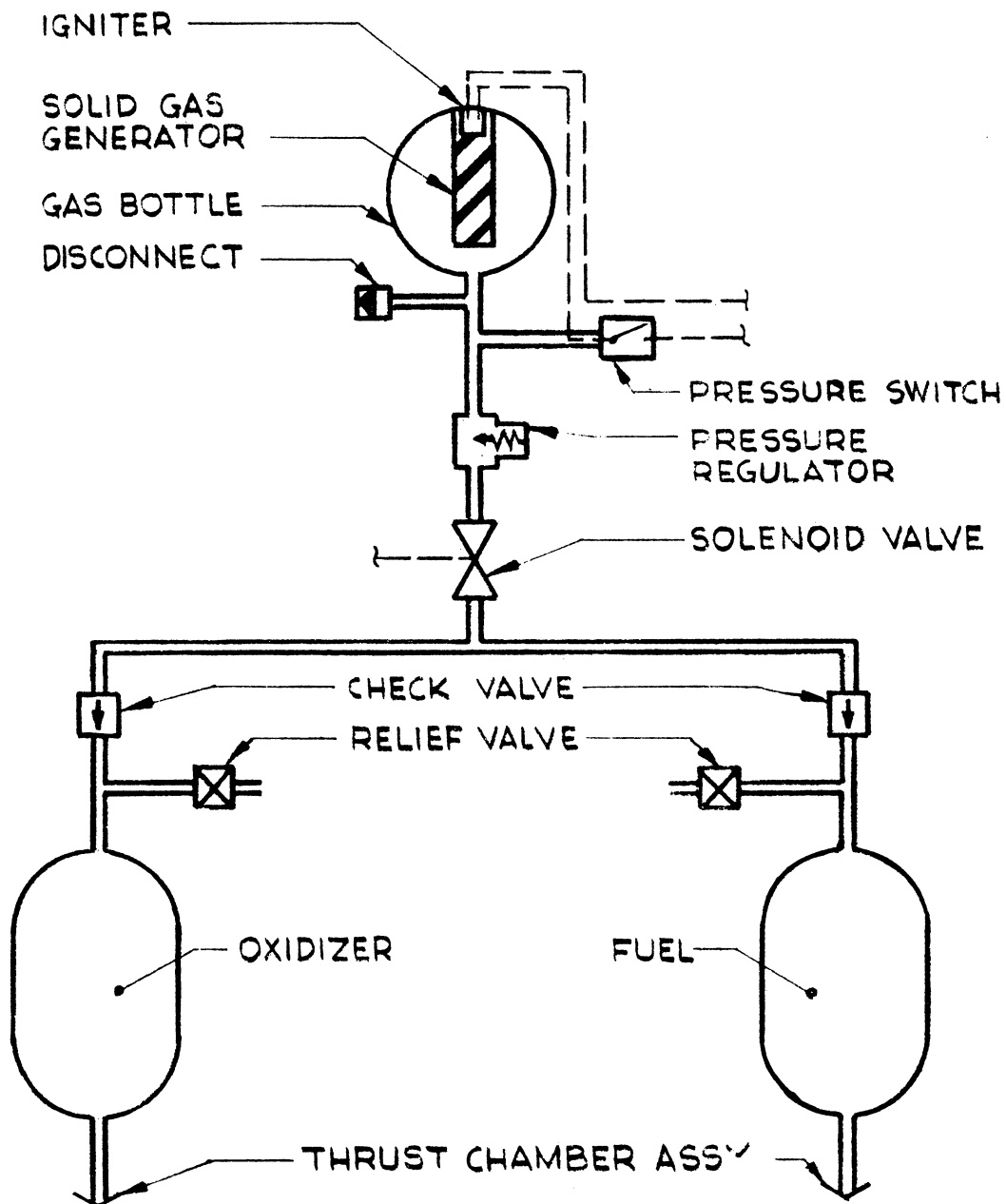
COMPONENT COMBINATION 13
JET PUMP



COMPONENT COMBINATION 14
STORED GAS (HEATED) AND FUEL-RICH
BI-PROPELLANT GAS GENERATOR



COMPONENT COMBINATION 15
STORED GAS (HEATED) AND
EVAPORATED FUEL



COMPONENT COMBINATION 16
STORED GAS
(SOLID PROPELLANT HEATED)

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LUNAR MISSION
PRESSURIZATION SYSTEMS INFLUENCE CURVES

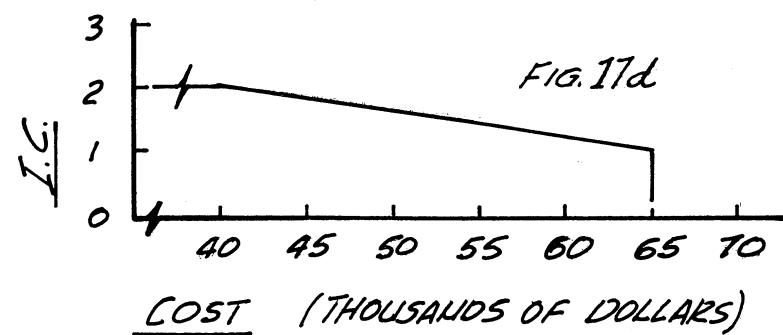
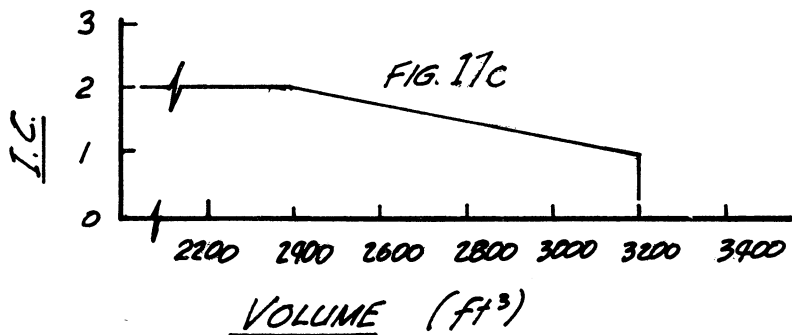
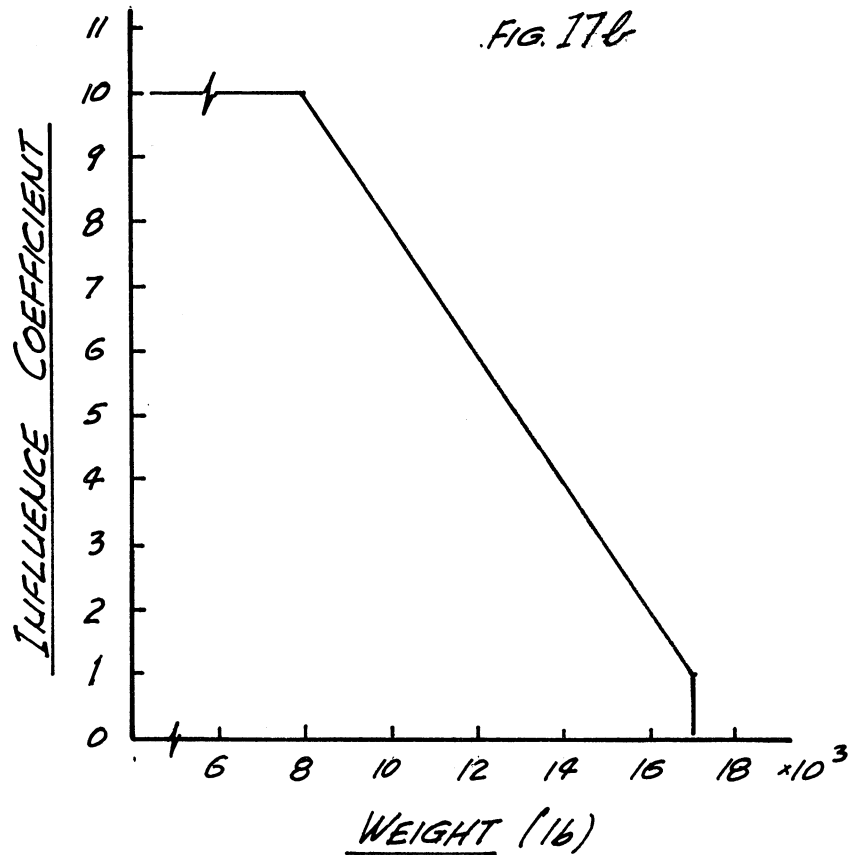
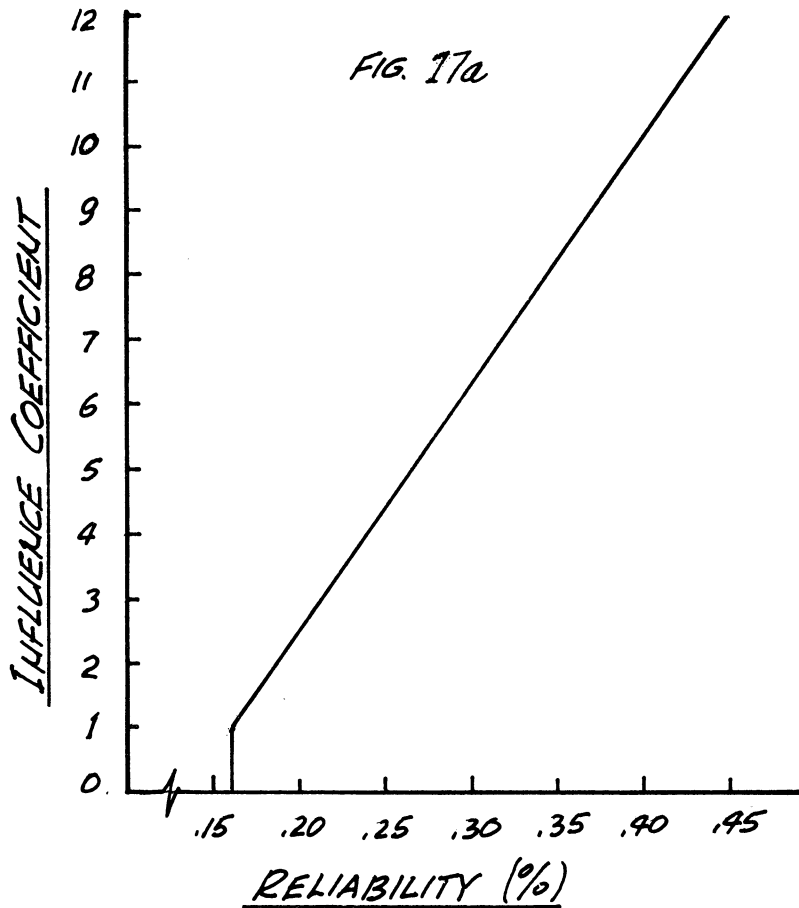


Figure 17

UNCLASSIFIED

UNCLASSIFIED

B_2H_6 LOADING SYSTEM AND IGNITION APPARATUS

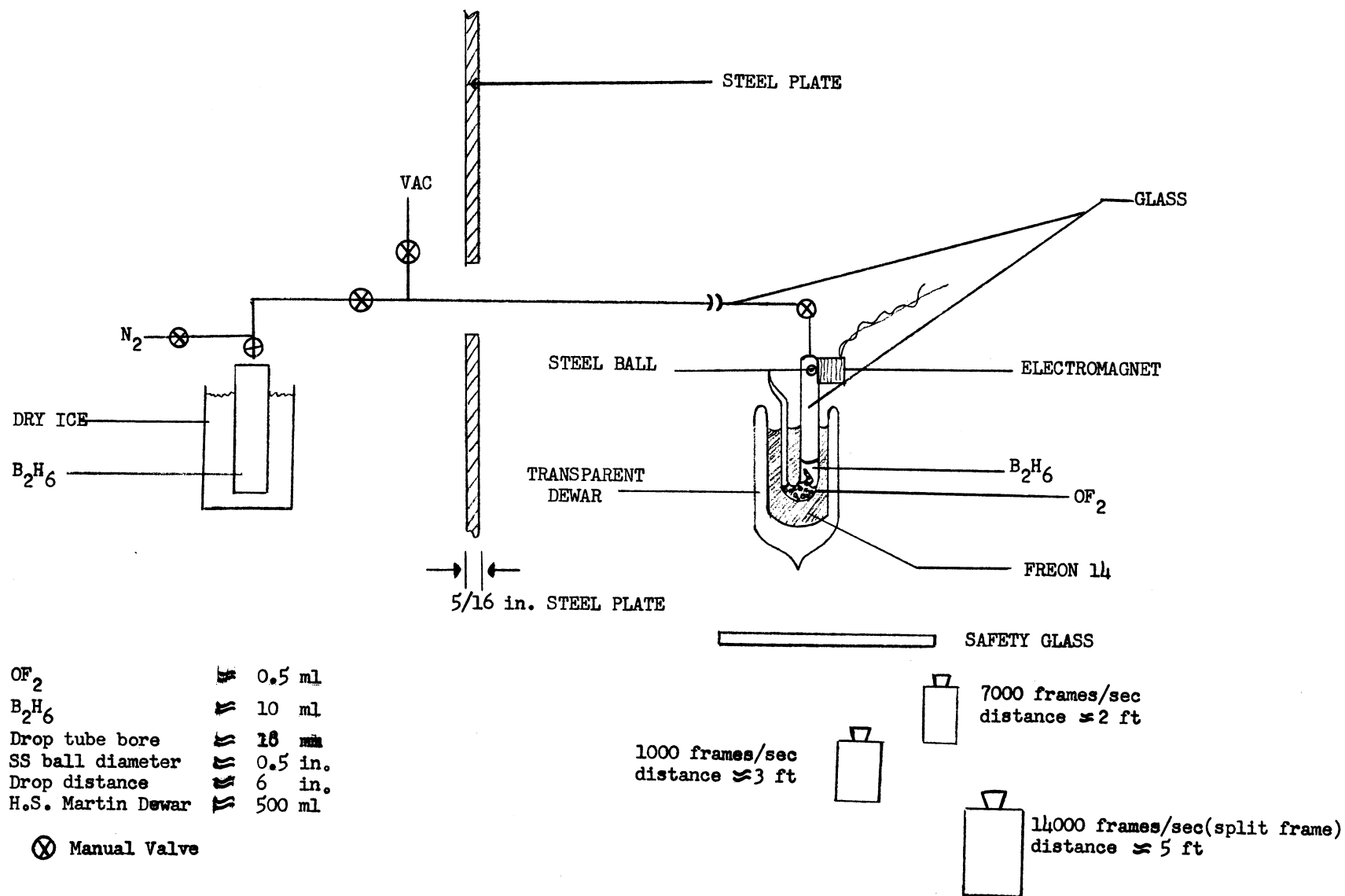
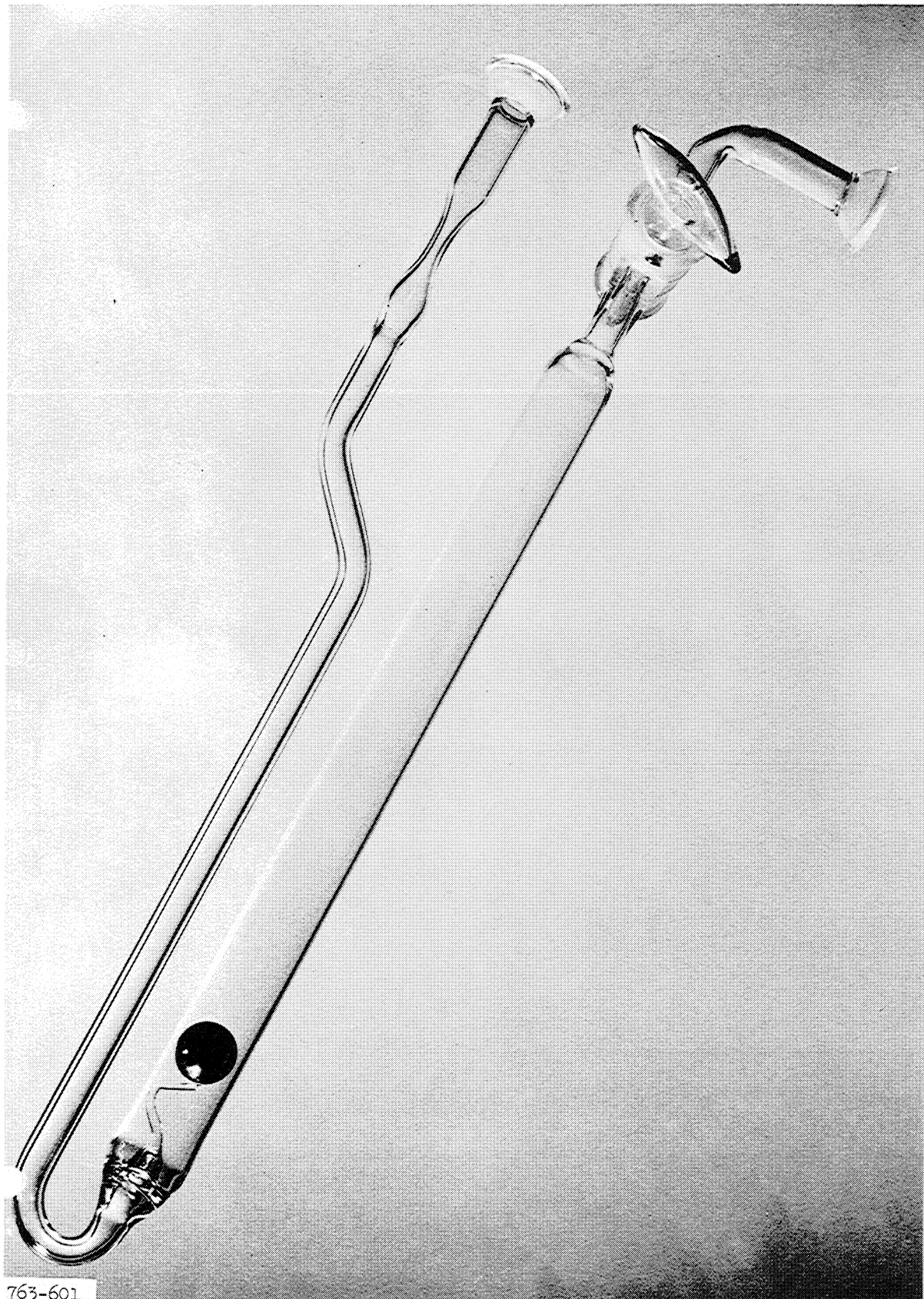


Figure 19

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763-601

PRESSURIZATION-TEST APPARATUS

FR = FLOW-RATE ORIFICE

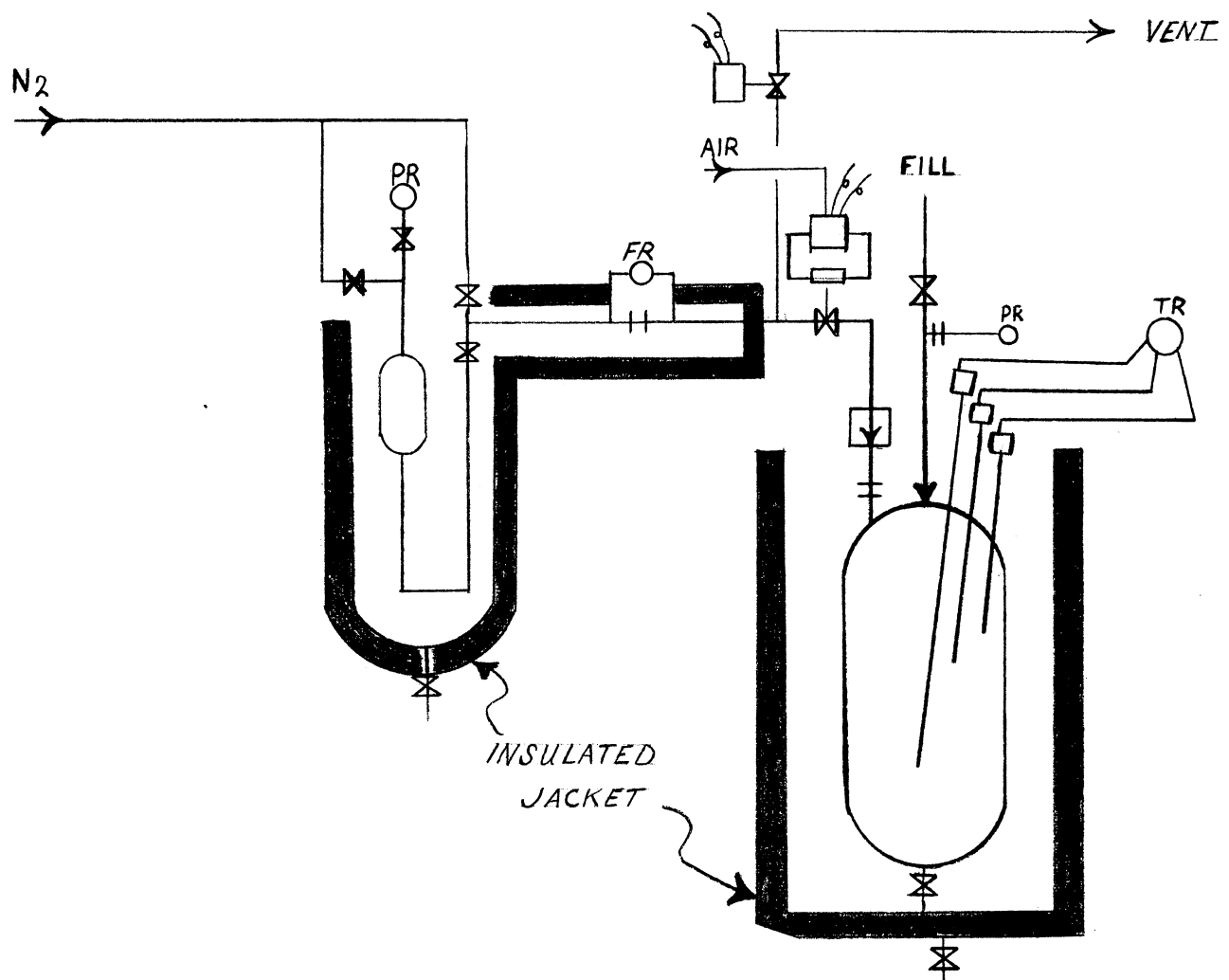
TR = TEMPERATURE REDUCER

PR = PRESSURE REDUCER

✕ = MANUAL VALVE

⦶ = ORIFICE

☐ = CHECK VALVE



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Report No. 2735

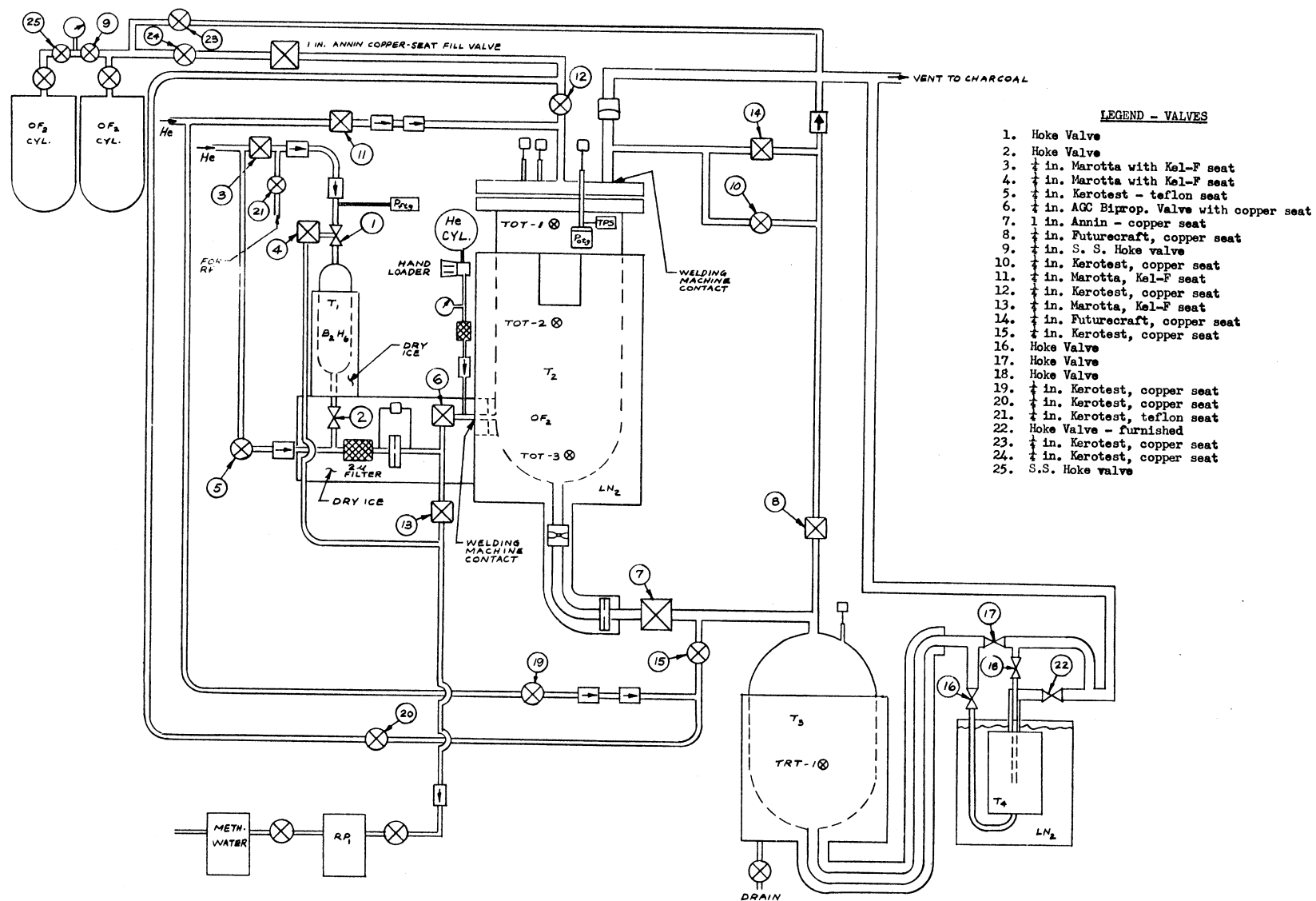
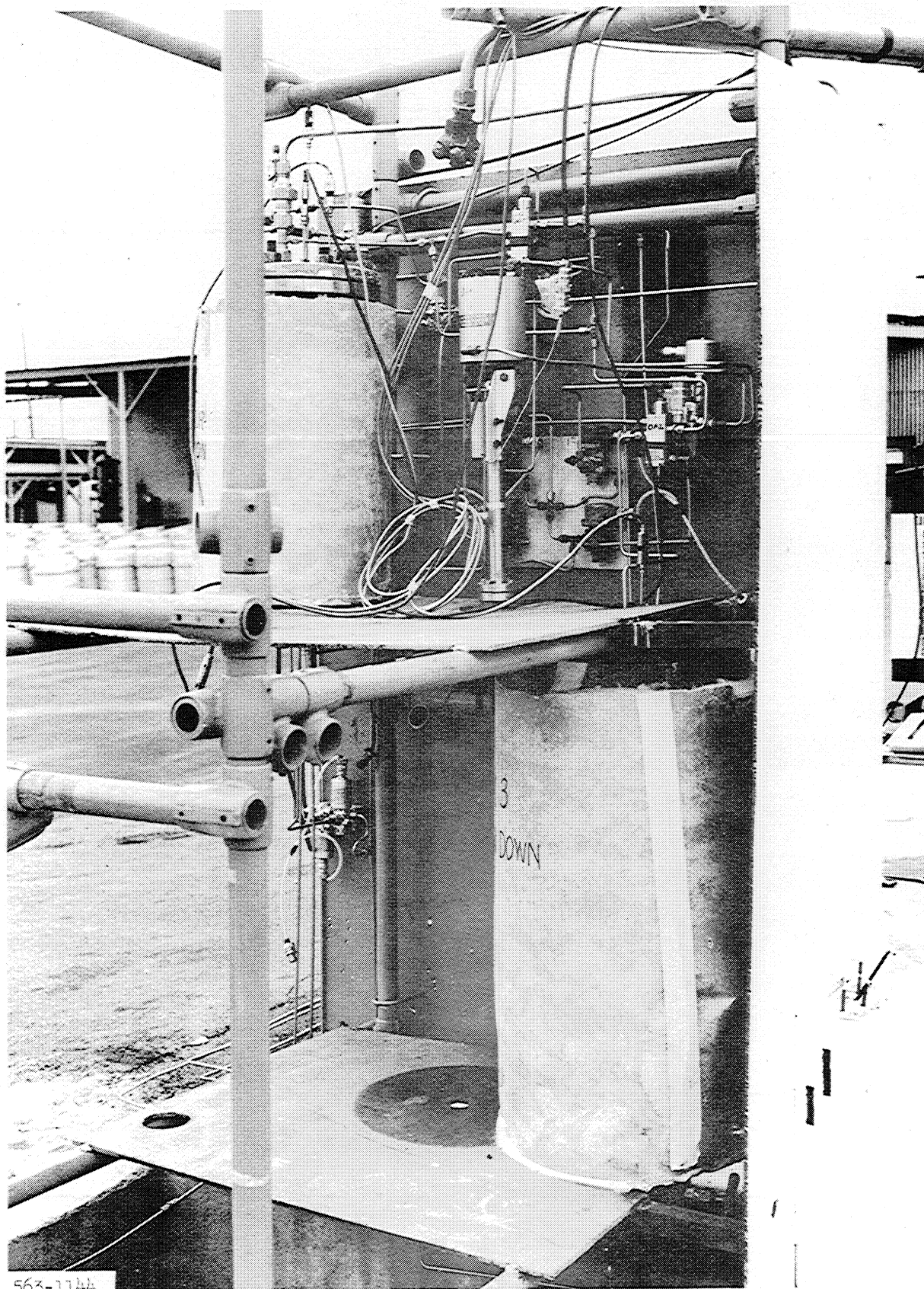


Figure 22

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Expulsion Test Set-Up

Figure 23

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Report No. 2735

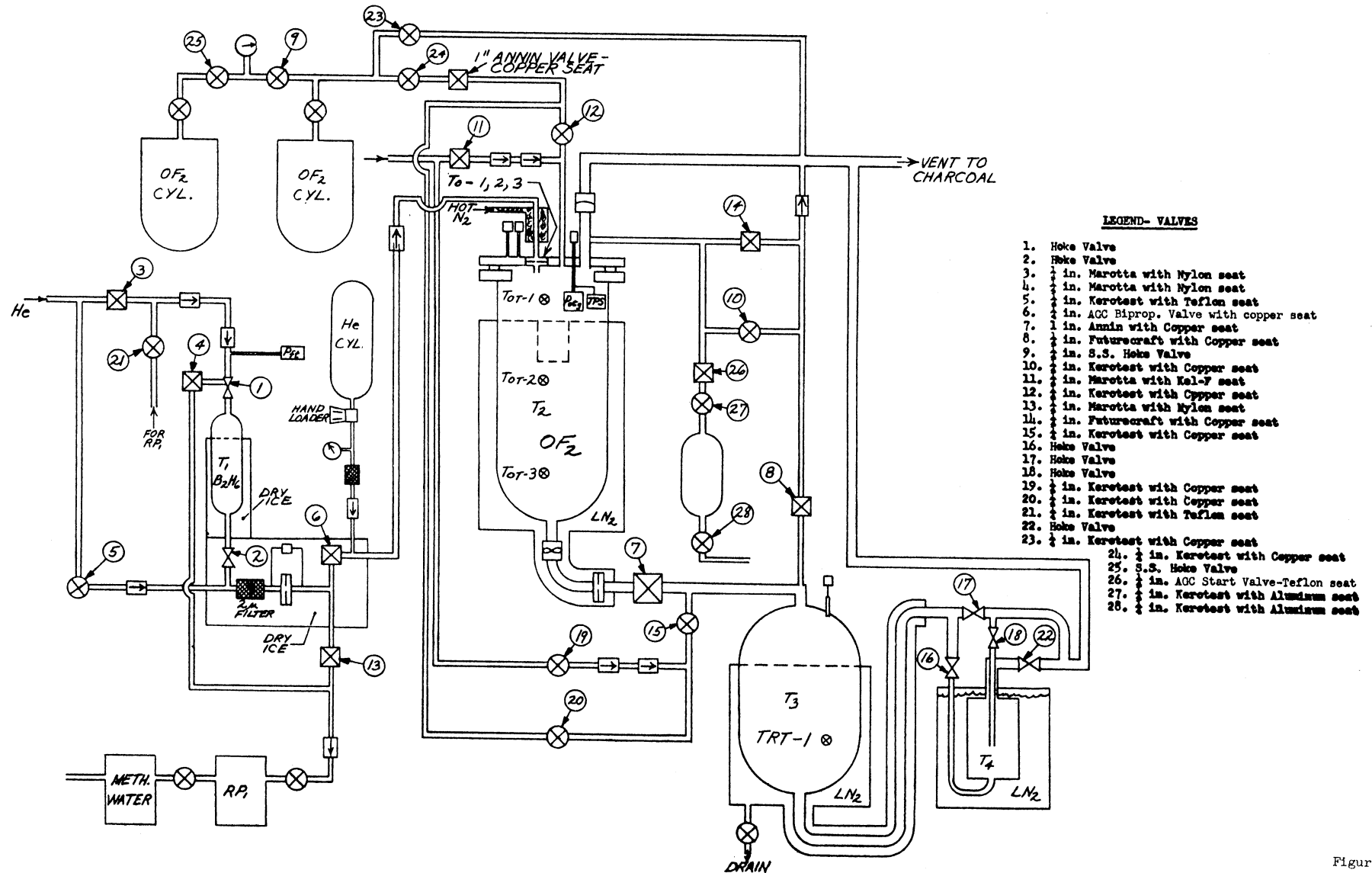
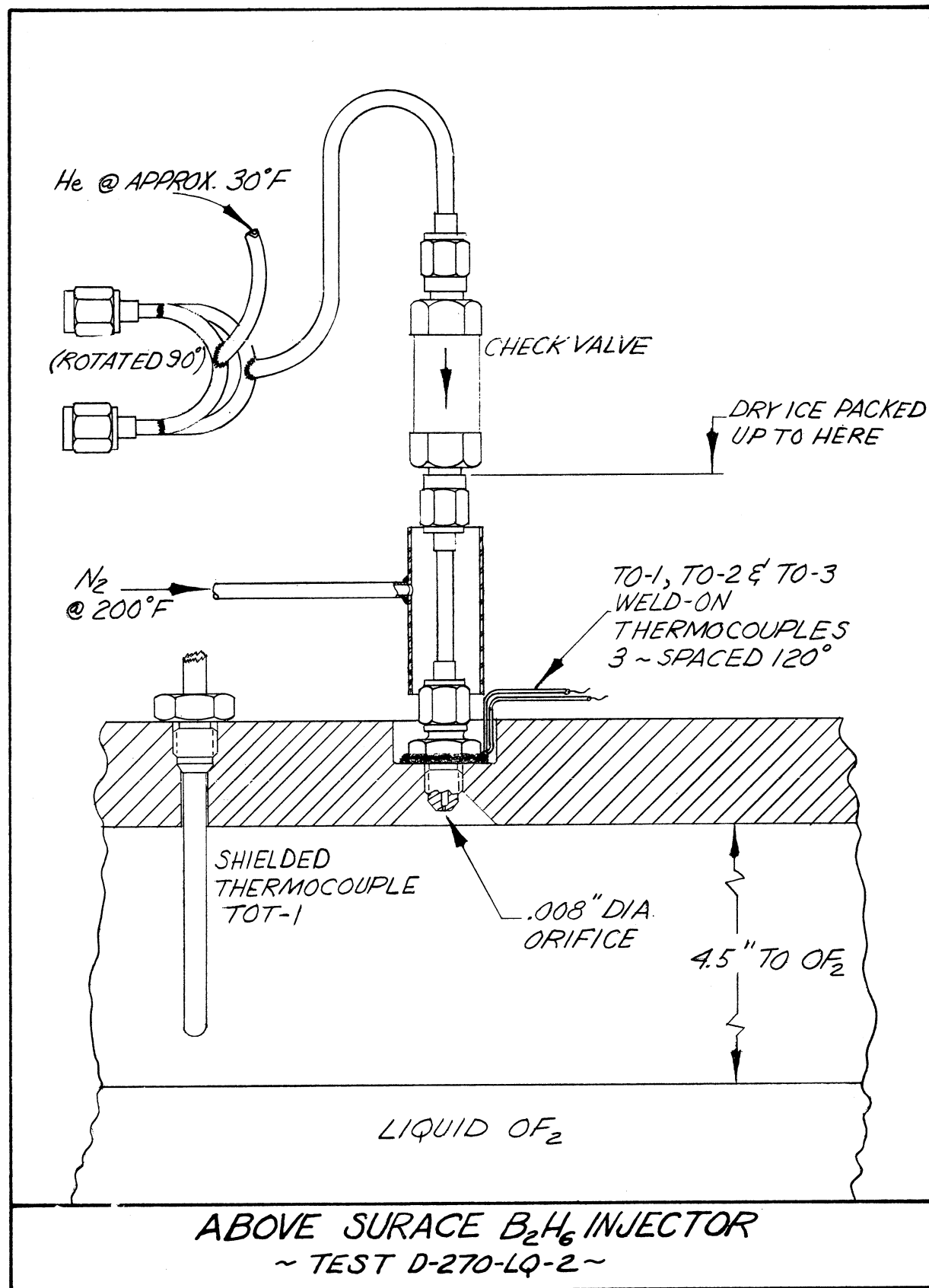
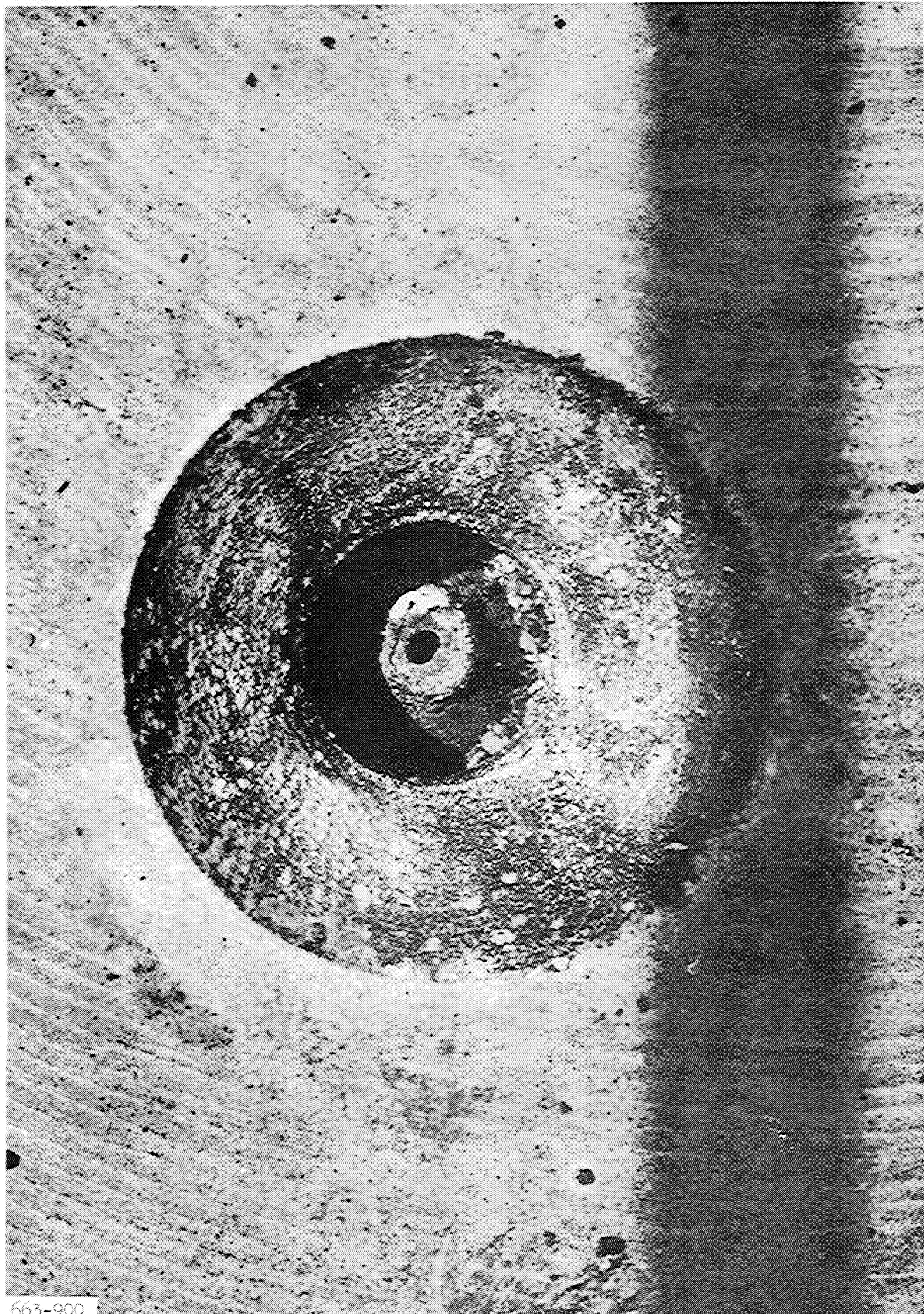


Figure 24

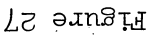
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Diborane Injector Orifice Following Test D 270-IQ-2

663-900



LIQUID INJECTION OF OF_2
INTO B_2H_6 - INJECTOR

MODIFIED SEAT
FITTING (FOR A
STANDARD COAXIAL
SOLENOID VALVE)

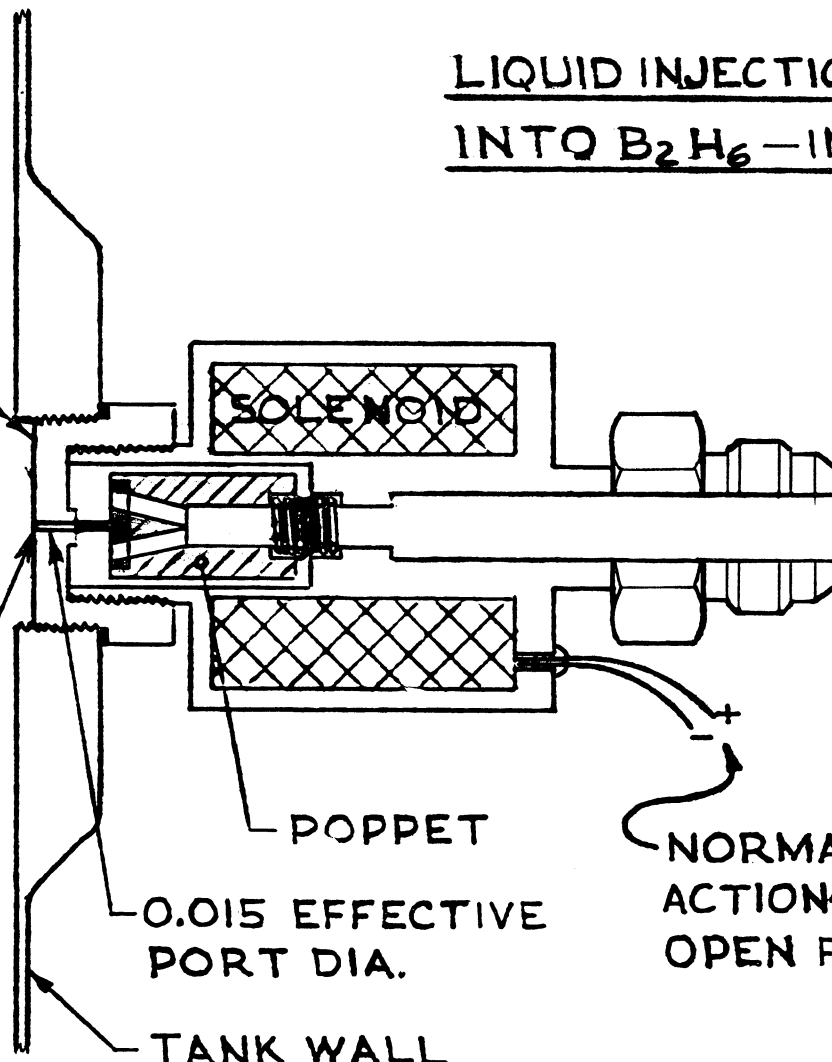
0.012 INJECTOR
PORT

POPPET

0.015 EFFECTIVE
PORT DIA.

TANK WALL

NORMALLY CLOSED
ACTION~POWER TO
OPEN FOR FLOW



UNCLASSIFIED

UNCLASSIFIED

